AEROSPACE REPORT NO.
ATR-2003(7949)-1

National Oceanic and Atmospheric Administration (NOAA) Geostationary Operational Environmental Satellite-R (GOES-R) Concept Design Center Space Segment Team Study

20 March 2003

Prepared by

THE CONCEPT DESIGN CENTER
Space Segment Team
Engineering and Technology Group

Prepared for

U.S. DEPT. OF COMMERCE, NOAA
Silver Spring, MD 20910

Contract No. 50-SPNA-0-00012

Civil & Commercial Division

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El Segundo, CA 90245-4691

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NATIONAL OCEANIC AND ATMOSPHERIC ADMINISTRATION (NOAA) GEOSTATIONARY OPERATIONAL ENVIRONMENTAL SATELLITE (GOES) BLOCK 5 CONCEPT DESIGN CENTER SPACE SEGMENT TEAM STUDY

Prepared

J. A. Aguilar
Vehicle Concepts Department
Systems Engineering Division

Approved

J. Soukup, Senior Project Leader
Reconnaissance Systems Division
NOAA/NESDIS Office of Systems
Development
Engineering and Technology Group

F. Knight, Director
Vehicle Concepts Department
Systems Engineering Division
Engineering and Technology Group

W. J. Hussey, Principal Director
Civil & Commercial Division

Abstract

The Aerospace Corporation's Concept Design Center Space Segment Team performed a Geostationary Operational Environment Satellite (GOES) study for the National Oceanic and Atmospheric Administration NOAA). During this study, thirteen spacecraft configurations were developed for the next-generation GOES. Three different architectures were explored with these spacecraft designs: (1) a consolidated spacecraft architecture, (2) a distributed spacecraft architecture, and (3) a MEO spacecraft architecture. This document contains the results of the study, including issues identified and recommendations.

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1. Introduction

Joseph Aguilar

1.1 Study Background

The National Oceanic and Atmospheric Administration (NOAA) Geostationary Operational Environment Satellite (GOES) Block 5 Concept Design Center (CDC) Space Segment Team (SST) study took place October 29–31, 2002 in The Aerospace Corporation's real-time design facility in El Segundo, California. A follow-up session was conducted on November 26. The purpose of the study was to assist NOAA in assessing the impact of various payload suites and architectures for the next-generation GOES spacecraft.

The Aerospace Corporation's CDC is a real-time design facility bringing together all of the subsystem experts needed to design a spacecraft. Each subsystem "seat" runs various software models to capture requirements, performance, and impact to the spacecraft. This information is then linked in real time to the other subsystems. Through an iterative process, the models will converge to a point design satisfying all of the given requirements for a particular configuration. Since the customer is present during these sessions, active discussion with the subsystem seats provides feedback to the customer, and allows the customer to clarify any requirements questions that may arise. It should be noted that, at the request of the customer, the cost, ground, and software subsystems were not used in this study.

For several weeks prior to the start of the design sessions, the study leads discussed spacecraft requirements with the customer, goals for the study, and pertinent configurations to be examined. Subsystem experts were brought in where necessary to lend expertise to the pre-session decisions and to perform any pre-work that would allow the sessions to run more smoothly.

1.2 Mission Overview

The GOES program is a key element in National Weather Service (NWS) operations, providing a continuous stream of environmental information (weather imagery and sounding data) used to support weather forecasting, severe-storm tracking, and meteorological research. From their geosynchronous positions over the eastern U.S./Atlantic and western U.S./Pacific, the two spacecraft can "stare" at most of the western hemisphere to provide cloud images, Earth surface temperatures, water-vapor fields, and vertical thermal and vapor structures. This data is used to follow the evolution of atmospheric phenomena, ensuring real-time coverage of short-lived dynamic events, especially severe local storms and tropical cyclones—two meteorological events that directly affect public safely, protection of property, and ultimately, economic health and development. – *GOES I-M DataBook*

1.3 Team Members

The CDC SST members that supported the GOES Block 5 design sessions are listed in Table 1.1. The team was formed from members of The Aerospace Corporation's technical staff who were selected to provide broad technical expertise and experience.

Table 1.1. CDC Team Members

Subsystem	Team Member
ADACS	Andrei Doran
Astrodynamics	Tom Lang, Laura Speckman
C&DH	Douglas Daughaday, Ron Selden
Configuration	Scott Szogas
Payload Communications	John O'Donnell
Power	Ed Berry
Propulsion	Trisha Beutien
Structures	Kenneth Mercer
Study Leads	Joseph Aguilar, Ron Bywater
Systems	Alice Moke
Thermal	Bill Fischer
TT&C	John O'Donnell

Hang-Kam Lee, from the Reliability and Statistics Office, provided some availability analysis to the customer during the study.

1.4 Customers

Mike Crison, Director of Requirements and Systems Programs in the Office of Systems Development at NOAA NESDIS, represented the interests of NOAA and was the primary customer for this study. There were a number of additional participants from NOAA, NASA, and other interested agencies. Jim Soukup, Senior Project Leader in the Reconnaissance Systems Division supporting NOAA NESDIS, was the Aerospace customer interface.

1.5 Disclaimer

This report constitutes the results of the CDC study. It is intended to assess feasibility and to estimate the required technologies, equipment, mass, and deployment strategy required to implement the customer's mission goals. The designs documented herein are intended to be conceptual solutions, developed with a minimum expenditure of work force and time. As a result, these representative solutions have not been optimized and may be incomplete or vary significantly from eventual systems. It is strongly recommended that a more detailed study be completed before final implementation decisions are made. Some more detailed analysis in the following areas is recommended:

- In-depth field-of-view analysis to verify that all of the sensors and antenna have adequate clear field-of-regard.

- Detailed trade study to determine hardware design and approach for payload and command and data handling interface.

- Detailed study of jitter sources, requirements, and solutions.

2. Payload

Joseph Aguilar

2.1 Payload Summary

There were several payloads used over the course of the 13 spacecraft configurations. Table 2.1 summarizes the payloads used for Configurations 1 to 6.

All of these payloads had a duty cycle of 100% in both daylight and eclipse with one exception. The SXI payload did not operate at all during eclipse. The SXI payload data also includes the solar coronagraph payload data. Table 2.2 summarizes changes and additions made to some of the payloads used for Configurations 7 to 13.

Table 2.1. Payload Summary for Configurations 1 to 6

Payload	Mass (kg)	Power (W)	Data Rate
ABI	220	410	21 Mbps
SXI	50	200	2.8 Mbps
GMS	300	300	500 kbps
Lightning Mapper	37.5	144	200 kbps
HES	157	527	65 Mbps
MFS	80	100	1.4 Mbps
SEM	54	94	560 Bps
FDS	180	190	1.2 Mbps
EHS	185	235	23 Mbps
DCS	17.9	29.7	
SAR	8.6	22.4	

Table 2.2. Payload Summary for Configurations 7 to 13

Payload	Mass (kg)	Power (W)	Data Rate
ABI	275	450	55 Mbps
HES	190	460	65 Mbps
Imaging Payload	150	150	500 kbps
A Sat Additional	61	100	-
B Sat Additional	100	150	-

2.2 Payload Acronyms

ABI	Advanced Baseline Imager
DCS	Data Collection System
EHS	Emissive Hyperspectral Sounder
FDS	Full Disk Sounder
GMS	Geostationary Microwave Sounder
HES	Hyperspectral Environmental Suite
MFS	Multi-function Sensor
SAR	Search and Rescue
SEM	Space Environment Monitor
SXI	Solar X-ray Imager

3. Systems

Alice Moke

3.1 Requirements

For this study, three different architectures were explored: (1) the ABC architecture, (2) the consolidated spacecraft architecture, and (3) the MEO spacecraft architecture. The A Sat, B Sat, and C Sat spacecraft were used for the ABC architecture. The AB Sat and MEO Sat space craft were used for the consolidated and MEO spacecraft architecture, respectively. All but one of the spacecraft configurations designed during this study were designed to operate in a Geosynchronous Earth Orbit (GEO). Table 3.1 shows the guidelines followed in designing these spacecraft.

One of the spacecraft configurations was designed to operate in a MEO and 19,000 km at 0° inclination. Its design requirements were the same as the others with the exception that it was designed to last 15 years. Additionally, there would be three operational spacecraft for this constellation plus one spare.

Table 3.1. GEO Spacecraft Guidelines

Mission	
Spacecraft Lifetime	10 years (7 operational, 2 on-orbit spare, 1 ground spare)
Ground Lifetime	16 years
Launch Date	2012
Technology Freeze Date	2008,
Mission Orbit	GEO, 75 West and 137 West
Inclination Tolerance	±0.5°
Desired Launch Vehicle	EELV Medium
Constellation Size	1 Spacecraft Cluster at Each Orbit Slot
Spacecraft	
Redundancy	Full
Heritage	Commercial
Stabilization	3-axis
Reposition Requirements	8 in lifetime (6 @ 1°/day, 2 @ 3°/day)
Slew Requirments	Bi-annual Yaw Flip
Knowledge	7 μrad Goal, 14 μrad Threshold
Pointing	150 μrad
Environment	Natural

In order to relate the technological maturity and technological risk to the uncertainty of the cost estimation elements, a numeric scale has been applied to many of the subsystems and components in the design. The numeric scale, referred to as the technology readiness level (TRL), was developed by NASA and has been adapted for use in this application. Figure 3.1 shows the relation between typical programmatic phases and the level of development that a technology has received. Note that in many cases the figure distinguishes between ground (G) and space (S) experience.

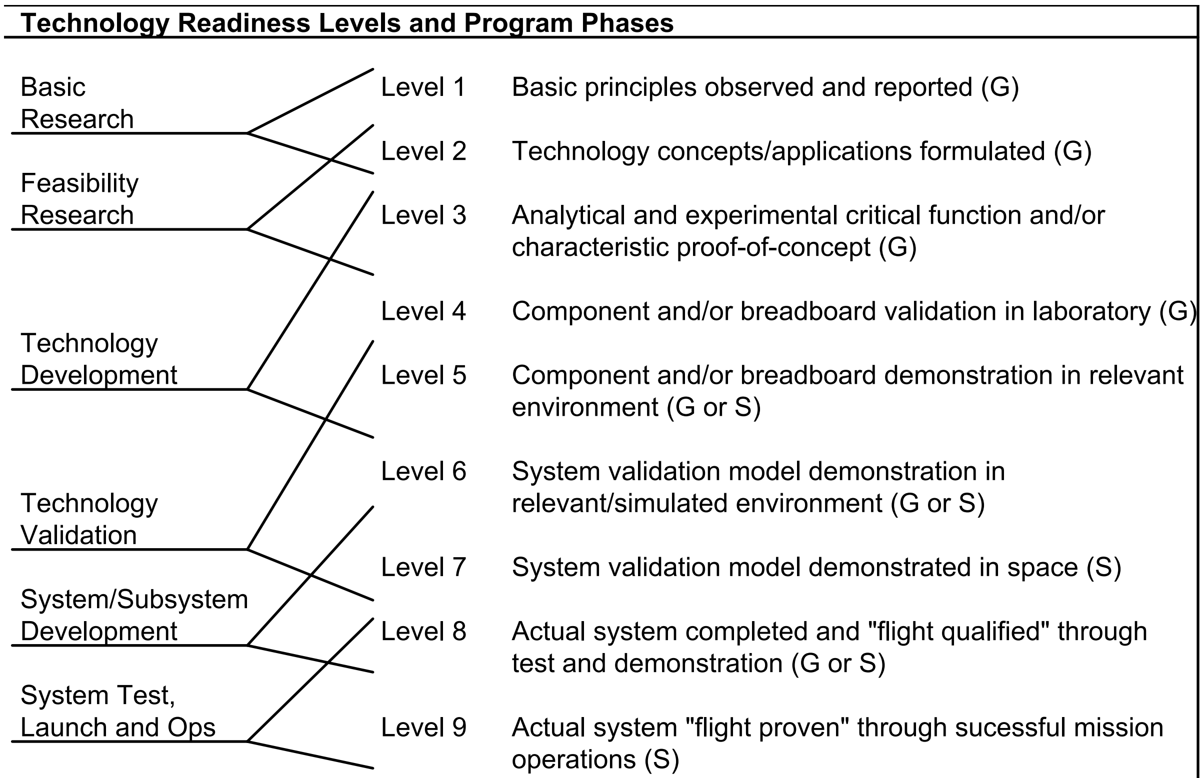

Figure 3.1. Technology readiness levels.

3.2 Spacecraft Configurations

Thirteen different spacecraft configurations were designed over the course of four design sessions. Multiple payload configurations were investigated during the course of the study. A brief description of each of the configurations is summarized in the following paragraphs. Mass and power results for each configuration are shown in Tables 3.2 to 3.14.

3.2.1 Configuration 1: A Sat–1st Run

This was the first spacecraft configuration to carry the ABI, SXI, Lightning Mapper, SAR, DCS, low-rate services, and global rebroadcast payloads. One A Sat and one B Sat spacecraft are launched together on an Atlas V 531 launch vehicle.

3.2.2 Configuration 2: B Sat–1st Run

This was the first spacecraft configuration to carry the HES, MFS, SEM, SAR, DCS, low-rate services, and global rebroadcast payloads. One A Sat and one B Sat spacecraft are launched together on an Atlas V 531 launch vehicle.

Table 3.2. Configuration 1: A Sat–1st Run

	Mass			Power [W]		NASA
	[kg]	[lbs]	%dry	Daylight	Eclipse	TRL
	Time (min) -->			1366.7	69.4	
Payload	**415.1**	**915.2**	**41%**	**1197.3**	**978.6**	
Custom Payload	334.0	736.5	34%	806.1	606.1	5
Payload Communications	64.9	143.0	7%	313.0	298.0	5
Payload Contingency	16.2	35.8		78.2	74.5	
Spacecraft	**722.3**	**1592.8**	**59%**	**480.7**	**480.7**	
Propulsion	111.7	246.3	11%	0.2	0.2	7
ADACS	55.0	121.3	6%	138.3	138.3	5
TT&C	16.8	37.1	2%	39.1	39.1	9
Command & Data Handling	11.2	24.7	1%	15.3	15.3	6
Thermal	28.2	62.2	3%	191.6	191.6	9
Power	161.5	356.1	17%	0.0	0.0	6
Structure	193.5	426.6	20%	0.0	0.0	6
Spacecraft Contingency	144.5	318.6		96.1	96.1	
Satellite Summary						
EOL Power				**1678.1**	**1459.3**	
BOL Power				**2469.7**		
Dry Mass	**1137.4**	**2508.0**				
Orbit Insertion Propellant	1359.2	2997.0				
On-Orbit Propellant	413.3	911.3				
Pressurant	4.2	9.2				
Wet Mass	**2914.0**	**6425.5**				

Table 3.3. Configuration 2: B Sat–1st Run

	Mass			Power [W]		NASA
	[kg]	[lbs]	%dry	Daylight	Eclipse	TRL
	Time (min) -->			1366.7	69.4	
Payload	**405.4**	**893.8**	**39%**	**1204.7**	**1204.7**	
Custom Payload	317.5	700.1	32%	773.1	773.1	5
Payload Communications	70.3	155.0	7%	345.3	345.3	5
Payload Contingency	17.6	38.7		86.3	86.3	
Spacecraft	**754.8**	**1664.3**	**61%**	**484.4**	**484.4**	
Propulsion	111.7	246.3	11%	0.2	0.2	7
ADACS	63.0	138.9	6%	138.3	138.3	5
TT&C	16.8	37.1	2%	39.1	39.1	9
Command & Data Handling	11.2	24.7	1%	15.3	15.3	6
Thermal	28.6	63.1	3%	194.6	194.6	9
Power	167.5	369.3	17%	0.0	0.0	6
Structure	205.0	452.0	21%	0.0	0.0	6
Spacecraft Contingency	151.0	332.9		96.9	96.9	
Satellite Summary						
EOL Power				**1689.1**	**1689.1**	
BOL Power				**2494.7**		
Dry Mass	**1160.1**	**2558.1**				
Orbit Insertion Propellant	1380.5	3044.1				
On-Orbit Propellant	420.1	926.2				
Pressurant	4.4	9.8				
Wet Mass	**2965.2**	**6538.2**				

Table 3.4. Configuration 3: B Sat Option 1

	Mass [kg]	Mass [lbs]	%dry	Power [W] Daylight	Power [W] Eclipse	NASA TRL
	Time (min) -->			1366.7	69.4	
Payload	**607.5**	**1339.5**	**46%**	**1063.9**	**1063.9**	
Custom Payload	525.5	1158.7	41%	671.1	671.1	5
Payload Communications	65.6	144.6	5%	314.2	314.2	5
Payload Contingency	16.4	36.2		78.6	78.6	
Spacecraft	**852.3**	**1879.3**	**54%**	**517.9**	**517.9**	
Propulsion	134.2	295.9	11%	0.2	0.2	7
ADACS	67.0	147.7	5%	138.3	138.3	5
TT&C	16.8	37.1	1%	39.1	39.1	9
Command & Data Handling	11.2	24.7	1%	15.3	15.3	6
Thermal	36.7	80.9	3%	221.3	221.3	9
Power	158.1	348.6	12%	0.0	0.0	6
Structure	257.9	568.6	20%	0.0	0.0	6
Spacecraft Contingency	170.5	375.9		103.6	103.6	
Satellite Summary						
EOL Power				**1581.7**	**1581.7**	
BOL Power				**2343.8**		
Dry Mass	**1459.8**	**3218.8**				
Orbit Insertion Propellant	1768.5	3899.6				
On-Orbit Propellant	537.5	1185.1				
Pressurant	5.7	12.6				
Wet Mass	**3771.5**	**8316.1**				

Table 3.5. Configuration 4: C Sat

	Mass [kg]	Mass [lbs]	%dry	Power [W] Daylight	Power [W] Eclipse	NASA TRL
	Time (min) -->			1366.7	69.4	
Payload	**357.8**	**789.0**	**43%**	**497.5**	**497.5**	
Custom Payload	337.5	744.2	41%	444.0	444.0	5
Payload Communications	16.3	35.9	2%	42.8	42.8	5
Payload Contingency	4.1	9.0		10.7	10.7	
Spacecraft	**576.5**	**1271.2**	**57%**	**387.9**	**387.9**	
Propulsion	96.8	213.4	12%	0.2	0.2	7
ADACS	55.0	121.3	7%	138.3	138.3	5
TT&C	16.8	37.1	2%	39.1	39.1	9
Command & Data Handling	11.2	24.7	1%	15.3	15.3	6
Thermal	23.5	51.9	3%	117.4	117.4	9
Power	93.0	205.1	11%	0.0	0.0	6
Structure	164.8	363.5	20%	0.0	0.0	6
Spacecraft Contingency	115.3	254.2		77.6	77.6	
Satellite Summary						
EOL Power				**885.4**	**885.4**	
BOL Power				**1320.5**		
Dry Mass	**934.3**	**2060.2**				
Orbit Insertion Propellant	1133.1	2498.4				
On-Orbit Propellant	343.6	757.6				
Pressurant	3.5	7.7				
Wet Mass	**2414.5**	**5323.9**				

Table 3.6. Configuration 5: Common Bus

	B Sat				B Sat Option 1			
	Mass [kg]	%dry	Daylight	Eclipse	Mass [kg]	%dry	Daylight	Eclipse
Payload	**405.4**	**39%**	**1204.7**	**1204.7**	**607.5**	**46%**	**1063.9**	**1063.9**
Custom Payload	317.5		773.1	773.1	525.5		671.1	671.1
Payload Thermal Control	0.0		0.0	0.0	0.0		0.0	0.0
Payload Processing	0.0		0.0	0.0	0.0		0.0	0.0
Payload Communications	70.3		345.3	345.3	65.6		314.2	314.2
Payload Contingency	17.6		86.3	86.3	16.4		78.6	78.6
Spacecraft	**754.8**		**484.4**	**484.4**	**852.3**		**517.9**	**517.9**
Propulsion	111.7	11%	0.2	0.2	134.2	11%	0.2	0.2
ADACS	63.0	6%	138.3	138.3	67.0	5%	138.3	138.3
TT&C	16.8	2%	39.1	39.1	16.8	1%	39.1	39.1
Command & Data Handling	11.2	1%	15.3	15.3	11.2	1%	15.3	15.3
Thermal	28.6	3%	194.6	194.6	36.7	3%	221.3	221.3
Power	167.5	17%	0.0	0.0	158.1	12%	0.0	0.0
Structure	205.0	21%	0.0	0.0	257.9	20%	0.0	0.0
Spacecraft Contingency	151.0		96.9	96.9	170.5		103.6	103.6
Satellite Summary								
EOL Power			**1689.1**	**1689.1**			**1581.7**	**1581.7**
BOL Power			**2494.7**				**2343.8**	
Dry Mass	**1160.1**				**1459.8**			
Orbit Insertion Propellant	1380.5	**33.3**			1768.5	**33.3**		
On-Orbit Propellant	420.1				537.5			
Pressurant	4.4				5.7			
Wet Mass	**2965.2**				**3771.5**			

Table 3.7. Configuration 6: AB Sat

	Mass			Power [W]		NASA TRL
	[kg]	[lbs]	%dry	Daylight	Eclipse	
	Time (min) -->			1366.7	69.4	
Payload	**717.2**	**1581.3**	**41%**	**1982.8**	**1764.0**	
Custom Payload	625.0	1378.1	37%	1527.1	1327.1	5
Payload Communications	73.7	162.6	4%	364.6	349.6	5
Payload Contingency	18.4	40.6		91.1	87.4	
Spacecraft	**1253.3**	**2763.4**	**59%**	**658.4**	**658.4**	
Propulsion	155.6	343.1	9%	0.2	0.2	7
ADACS	87.0	191.8	5%	138.3	138.3	5
TT&C	16.8	37.1	1%	39.1	39.1	9
Command & Data Handling	11.2	24.7	1%	15.3	15.3	6
Thermal	60.2	132.7	4%	333.8	333.8	9
Power	254.1	560.2	15%	0.0	0.0	6
Structure	417.8	921.2	25%	0.0	0.0	6
Spacecraft Contingency	250.7	552.7		131.7	131.7	
Satellite Summary						
EOL Power				**2641.2**	**2422.5**	
BOL Power				**3985.1**		
Dry Mass	**1970.4**	**4344.8**				
Orbit Insertion Propellant	2367.6	5220.5				
On-Orbit Propellant	719.5	1586.6				
Pressurant	7.6	16.8				
Wet Mass	**5065.2**	**11168.7**				

Table 3.8. Configuration 7: A Sat–2nd Run

	Mass			Power [W]		NASA
	[kg]	[lbs]	%dry	Daylight	Eclipse	TRL
	Time (min) -->			1366.7	69.4	
Payload	**475.6**	**1048.6**	**43%**	**1270.5**	**1070.5**	
Custom Payload	389.0	857.7	37%	846.1	646.1	5
Payload Communications	69.2	152.7	7%	339.6	339.6	5
Payload Contingency	17.3	38.2		84.9	84.9	
Spacecraft	**757.9**	**1671.2**	**57%**	**502.3**	**502.3**	
Propulsion	111.7	246.3	10%	0.2	0.2	7
ADACS	55.0	121.3	5%	138.3	138.3	5
TT&C	16.8	37.1	2%	39.1	39.1	9
Command & Data Handling	11.2	24.7	1%	15.3	15.3	6
Thermal	30.7	67.8	3%	208.9	208.9	9
Power	170.5	376.0	16%	0.0	0.0	6
Structure	210.4	463.8	20%	0.0	0.0	7
Spacecraft Contingency	151.6	334.2		100.5	100.5	
Satellite Summary						
EOL Power				**1772.8**	**1572.8**	
BOL Power				**2610.0**		
Dry Mass	**1233.5**	**2719.8**				
Orbit Insertion Propellant	1481.3	3266.3				
On-Orbit Propellant	450.3	992.9				
Pressurant	4.5	10.0				
Wet Mass	**3169.6**	**6989.1**				

Table 3.9. Configuration 8: B Sat–2nd Run

	Mass			Power [W]		NASA
	[kg]	[lbs]	%dry	Daylight	Eclipse	TRL
	Time (min) -->			1366.7	69.4	
Payload	**438.4**	**966.6**	**41%**	**1137.7**	**1137.7**	
Custom Payload	350.5	772.9	34%	706.1	706.1	5
Payload Communications	70.3	155.0	7%	345.3	345.3	5
Payload Contingency	17.6	38.7		86.3	86.3	
Spacecraft	**756.4**	**1667.8**	**59%**	**492.8**	**492.8**	
Propulsion	111.7	246.3	11%	0.2	0.2	7
ADACS	63.0	138.9	6%	138.3	138.3	5
TT&C	16.8	37.1	2%	39.1	39.1	9
Command & Data Handling	11.2	24.7	1%	15.3	15.3	6
Thermal	29.6	65.3	3%	201.3	201.3	9
Power	161.3	355.6	16%	0.0	0.0	6
Structure	211.5	466.4	21%	0.0	0.0	6
Spacecraft Contingency	151.3	333.6		98.6	98.6	
Satellite Summary						
EOL Power				**1630.5**	**1630.5**	
BOL Power				**2394.8**		
Dry Mass	**1194.7**	**2634.4**				
Orbit Insertion Propellant	1427.8	3148.2				
On-Orbit Propellant	434.1	957.1				
Pressurant	4.6	10.1				
Wet Mass	**3061.2**	**6749.9**				

Table 3.10. Configuration 9: A Sat without Low-rate Services and GRB

	Mass			Power [W]		NASA TRL
	[kg]	[lbs]	%dry	Daylight	Eclipse	
	Time (min) -->			1366.7	69.4	
Payload	**408.1**	**899.9**	**43%**	**883.8**	**683.8**	
Custom Payload	362.5	799.3	39%	794.0	594.0	5
Payload Communications	36.5	80.5	4%	71.8	71.8	5
Payload Contingency	9.1	20.1		18.0	18.0	
Spacecraft	**657.8**	**1450.4**	**57%**	**468.1**	**468.1**	
Propulsion	105.5	232.7	11%	0.2	0.2	7
ADACS	49.0	108.0	5%	138.3	138.3	5
TT&C	16.8	37.1	2%	39.1	39.1	9
Command & Data Handling	11.2	24.7	1%	15.3	15.3	6
Thermal	26.7	58.9	3%	181.5	181.5	9
Power	134.6	296.8	15%	0.0	0.0	6
Structure	182.4	402.1	20%	0.0	0.0	7
Spacecraft Contingency	131.6	290.1		93.6	93.6	
Satellite Summary						
EOL Power				**1351.9**	**1151.9**	
BOL Power				**2034.9**		
Dry Mass	**1065.9**	**2350.3**				
Orbit Insertion Propellant	1285.2	2833.9				
On-Orbit Propellant	388.6	857.0				
Pressurant	3.9	8.7				
Wet Mass	**2743.7**	**6049.8**				

Table 3.11. Configuration 10: B Sat without Low-rate Services and GRB

	Mass			Power [W]		NASA TRL
	[kg]	[lbs]	%dry	Daylight	Eclipse	
	Time (min) -->			1366.7	69.4	
Payload	**370.9**	**817.9**	**41%**	**750.9**	**750.9**	
Custom Payload	324.0	714.4	37%	654.0	654.0	5
Payload Communications	37.5	82.8	4%	77.5	77.5	5
Payload Contingency	9.4	20.7		19.4	19.4	
Spacecraft	**644.7**	**1421.6**	**59%**	**456.4**	**456.4**	
Propulsion	100.8	222.2	11%	0.2	0.2	7
ADACS	55.0	121.3	6%	138.3	138.3	5
TT&C	16.8	37.1	2%	39.1	39.1	9
Command & Data Handling	11.2	24.7	1%	15.3	15.3	6
Thermal	25.3	55.9	3%	172.2	172.2	9
Power	124.9	275.4	14%	0.0	0.0	6
Structure	181.7	400.8	21%	0.0	0.0	7
Spacecraft Contingency	128.9	284.3		91.3	91.3	
Satellite Summary						
EOL Power				**1207.3**	**1207.3**	
BOL Power				**1816.7**		
Dry Mass	**1015.6**	**2239.4**				
Orbit Insertion Propellant	1220.8	2692.0				
On-Orbit Propellant	371.2	818.4				
Pressurant	3.7	8.3				
Wet Mass	**2611.4**	**5758.1**				

Table 3.12. Configuration 11: MEO Sat

	Mass		%dry	Power [W]		NASA TRL
	[kg]	[lbs]		Daylight	Eclipse	
	Time (min) -->			616.4	54.2	
Payload	**239.2**	**527.4**	**37%**	**499.3**	**499.3**	
Custom Payload	176.5	389.2	29%	202.1	202.1	5
Payload Communications	50.2	110.6	8%	237.8	237.8	5
Payload Contingency	12.5	27.6		59.4	59.4	
Spacecraft	**481.4**	**1061.5**	**63%**	**336.2**	**336.2**	
Propulsion	89.8	198.1	15%	0.5	0.5	7
ADACS	32.4	71.4	5%	114.3	114.3	5
TT&C	19.5	42.9	3%	21.9	21.9	9
Command & Data Handling	13.3	29.3	2%	15.3	15.3	6
Thermal	17.7	39.0	3%	116.9	116.9	9
Power	89.0	196.3	15%	0.0	0.0	6
Structure	123.4	272.2	20%	0.0	0.0	7
Spacecraft Contingency	96.3	212.3		67.2	67.2	
Satellite Summary						
EOL Power				**835.5**	**835.5**	
BOL Power				**1525.5**		
Dry Mass	**720.6**	**1588.9**				
Orbit Insertion Propellant	857.9	1891.6				
On-Orbit Propellant	60.8	134.0				
Pressurant	2.6	5.7				
Wet Mass	**1641.8**	**3620.2**				

Table 3.13. Configuration 12: A Sat with Additional Payload Mass and Power

	Mass		%dry	Power [W]		NASA TRL
	[kg]	[lbs]		Daylight	Eclipse	
	Time (min) -->			1366.7	69.4	
Payload	**536.6**	**1183.1**	**44%**	**1370.5**	**1170.5**	
Custom Payload	450.0	992.3	38%	946.1	746.1	5
Payload Communications	69.2	152.7	6%	339.6	339.6	5
Payload Contingency	17.3	38.2		84.9	84.9	
Spacecraft	**841.3**	**1855.0**	**56%**	**533.6**	**533.6**	
Propulsion	135.6	299.1	11%	0.2	0.2	7
ADACS	57.0	125.7	5%	138.3	138.3	5
TT&C	16.8	37.1	1%	39.1	39.1	9
Command & Data Handling	11.2	24.7	1%	15.3	15.3	6
Thermal	34.4	75.9	3%	233.9	233.9	9
Power	183.3	404.2	15%	0.0	0.0	6
Structure	234.6	517.3	20%	0.0	0.0	7
Spacecraft Contingency	168.3	371.0		106.7	106.7	
Satellite Summary						
EOL Power				**1904.1**	**1704.1**	
BOL Power				**2817.2**		
Dry Mass	**1377.8**	**3038.1**				
Orbit Insertion Propellant	1658.9	3657.9				
On-Orbit Propellant	504.2	1111.9				
Pressurant	5.1	11.2				
Wet Mass	**3546.1**	**7819.1**				

Table 3.14. Configuration 13: B Sat with Additional Payload Mass and Power

	Mass			Power [W]		NASA TRL
	[kg]	[lbs]	%dry	Daylight	Eclipse	
	Time (min) -->			1366.7	69.4	
Payload	**538.4**	**1187.1**	**43%**	**1287.7**	**1287.7**	
Custom Payload	450.5	993.4	37%	856.1	856.1	5
Payload Communications	70.3	155.0	6%	345.3	345.3	5
Payload Contingency	17.6	38.7		86.3	86.3	
Spacecraft	**860.3**	**1896.9**	**57%**	**537.7**	**537.7**	
Propulsion	135.6	299.1	11%	0.2	0.2	7
ADACS	63.0	138.9	5%	138.3	138.3	5
TT&C	16.8	37.1	1%	39.1	39.1	9
Command & Data Handling	11.2	24.7	1%	15.3	15.3	6
Thermal	34.9	77.0	3%	237.2	237.2	9
Power	180.4	397.7	15%	0.0	0.0	6
Structure	246.3	543.1	20%	0.0	0.0	6
Spacecraft Contingency	172.1	379.4		107.5	107.5	
Satellite Summary						
EOL Power				**1825.4**	**1825.4**	
BOL Power				**2702.7**		
Dry Mass	**1398.6**	**3084.0**				
Orbit Insertion Propellant	1682.4	3709.6				
On-Orbit Propellant	511.3	1127.3				
Pressurant	5.4	11.9				
Wet Mass	**3597.7**	**7932.9**				

3.2.3 Configuration 3: B Sat Option 1

This spacecraft configuration is identical to Configuration 2 with the following exception. Instead of carrying the HES payload, the FDS and EHS payloads were used.

3.2.4 Configuration 4: C Sat

This spacecraft carried the GMS and Lightning Mapper payloads. Two of these spacecraft are launched together on an Atlas V 521.

3.2.5 Configuration 5: Common Bus

Each of the first four configurations were analyzed to determine whether one bus could serve as a common bus for all. For A Sat, B Sat, and C Sat, the B Sat bus represents the common bus design. Each spacecraft subsystem of the B Sat can accommodate the A Sat and C Sat spacecraft requirements. For A Sat, B Sat Option 1, and C Sat, the B Sat Option 1 bus represents the common bus design. Again, each spacecraft subsystem of the B Sat Option 1 can accommodate the A Sat and C Sat spacecraft requirements. However, this bus is substantially over-designed for the A Sat and C Sat payload suites.

3.2.6 Configuration 6: AB Sat

The AB spacecraft combined the functionality of the A and B spacecraft into a single spacecraft. As such, it carried the ABI, SXI, Lighting Mapper, HES, MFS, SEM, SAR, DCS, low-rate services, and global rebroadcast payloads. A single spacecraft is launched on a Delta V M+ (5,4) launch vehicle.

3.2.7 Configuration 7: A Sat–2nd Run

This was the second spacecraft configuration to carry the ABI, SXI, Lightning Mapper, SAR, DCS, low-rate services, and global rebroadcast payloads. One A Sat-2nd Run and one B Sat-2nd Run spacecraft are launched together on an Atlas V 541 launch vehicle.

3.2.8 Configuration 8: B Sat–2nd Run

This was the second spacecraft configuration to carry the HES, MFS, SEM, SAR, DCS, low-rate services, and global rebroadcast payloads. One A Sat-2nd Run and one B Sat-2nd Run spacecraft are launched together on an Atlas V 541 launch vehicle.

3.2.9 Configuration 9: A Sat Without Low-rate Services and GRB

This spacecraft is identical to Configuration 7 except that the low-rate services, GRB, SAR, and DCS payloads have been removed. One A Sat without Low-Rate Services and GRB and one B Sat without Low-rate Services and GRB spacecraft are launched together on an Atlas V 531 launch vehicle.

3.2.10 Configuration 10: B Sat Without Low-rate Services and GRB

This spacecraft is identical to Configuration 8 except that the low-rate services, GRB, SAR, and DCS payloads have been removed. One A Sat without Low-Rate Services and GRB and one B Sat without Low-rate Services and GRB spacecraft are launched together on an Atlas V 531 launch vehicle.

3.2.11 Configuration 11: MEO Sat

This spacecraft carries the low-rate services, GRB, SAR, DCS, and an imaging payload. Two of these spacecraft are launched together on a Delta IV M+ 5,2.

3.2.12 Configuration 12: A Sat with Additional Payload Mass and Power

This was the third spacecraft configuration to carry the ABI, SXI, Lightning Mapper, SAR, DCS, low-rate services, and global rebroadcast payloads. This configuration represents the baseline A Sat spacecraft. One A Sat with Additional Payload Mass and Power and one B Sat with Additional Payload Mass and Power spacecraft are launched together on an Atlas V 551 launch vehicle.

3.2.13 Configuration 13: B Sat with Additional Payload Mass and Power

This was the third spacecraft configuration to carry the HES, MFS, SEM, SAR, DCS, low-rate services, and global rebroadcast payloads. This configuration represents the baseline B Sat spacecraft.

One A Sat with Additional Payload Mass and Power and one B Sat with Additional Payload Mass and Power spacecraft are launched together on an Atlas V 551 launch vehicle.

3.3 Configurations Summary

Of the 13 spacecraft configurations generated, the configurations fell into six different designs. They were A Sat, B Sat, C Sat, Common Bus, AB Sat, and MEO Sat. For the A, B, and C Sats or ABC Architecture, Configuration 12 and 13 represent the baseline A and B spacecraft. Configurations 1, 2, 7, and 8 are older versions of these baseline spacecraft. Configurations 9 and 10 are essentially the same as the baseline spacecraft with the exception of removing some communications.

The Common Bus will not be discussed further since the payloads carried on the common bus were updated since that design. Table 3.15 shows a top-level summary of the other 5 spacecraft configurations.

The spacecraft listed in Table 3.15 will be discussed in greater detail in each of the subsystem sections of this report.

Table 3.15. Spacecraft Configurations Summary

	A Sat	B Sat	C Sat	AB Sat	MEO Sat
Configuration	12	13	4	6	11
Payload Mass (kg)	536.6	538.4	357.8	717.2	239.2
Payload Power (W)	1370.5	1287.7	497.5	1982.8	499.3
Spacecraft Mass (kg)	841.3	860.3	576.5	1253.3	481.4
Spacecraft Power (W)	533.6	537.7	387.9	658.4	336.2
BOL Power (W)	2817.2	2702.7	1320.5	3985.1	1525.5
Dry Mass (kg)	1377.8	1398.6	934.3	1970.4	720.6
Wet Mass (kg)	3546.1	3597.7	2414.5	5065.2	1641.8
Launch Mass (kg)	7658.0		6128.0	5065.2	3717.1
Launched with:	B Sat	A Sat	C Sat	-	MEO Sat
Launch Vehicle	Atlas V 551		Atlas V 521	Delta IV (5,4)	Delta IV (5,2)
Launch Margin (kg)	857.3		864.4	1346.8	5854.0

3.4 Common Spacecraft Configuration Observations

For all of the spacecraft configurations, there were no additional mass and power margins placed on the payloads at the customer’s direction with the exception of the communications payload. Mass and power margins were carried for the spacecraft bus and communications payload, which was 25%.

All of the spacecraft were designed to be injected into a transfer orbit. At that transfer orbit, the on-board propulsion system would circularize the spacecraft orbit to either GEO or MEO. All of the spacecraft were launched by either the Delta or Atlas EELVs. All of the spacecraft configurations, with one exception, were dual manifested with another spacecraft. Table 3.15 shows the launch vehicle used along with the launch vehicle margin.

4. Configuration

Scott Szogas

4.1 Overview

The CDC routinely generates a conceptual-level satellite configuration (three-dimensional model) in order to provide geometric information about the satellite being studied. The configuration is built up from the primary building blocks of the satellite, including the spacecraft bus, solar arrays, sensors, and antennas. The structural subsystem expert, using historical satellite data, determines the volume for the spacecraft bus. The dimensions of the satellite are chosen to provide the proper satellite bus volume. The sensors are constructed based on customer-provided geometry. The antenna dimensions and configurations are provided by the communications subsystem expert based on the mission requirements.

The spacecraft configuration model is constructed to provide a geometric baseline for the satellite. The sizes of the array and bus structure are coordinated with the power and structure subsystems during the study. The geometric model is also used to generate moments of inertia of the satellite based on the locations of the various components being modeled. The attitude determination and control subsystem expert uses the mass moment of inertia to determine the actuators required for controlling the satellite attitude. Radiator areas and locations are also discussed with the thermal expert to assure that adequate thermal dissipation is present.

4.2 Analysis

The deployed on-orbit configurations of the five principal spacecraft generated during the study sessions are presented in Figures 4.1 through 4.5. In general, the payload complement presented in these configurations was accommodated without much difficulty. The AB Sat configuration was the most stressing as far as utilization of the nadir panel mounting surface, and required that the nadir panel be placed vertically during launch. A single-axis articulated solar array was chosen to provide an unobstructed field of view for the sensor radiators. An in-depth field-of-view analysis should be performed for the configurations in order to verify that the sensors and antenna all have adequate clear field-of-regard.

Figures 4.6 through 4.9 show notional launch configurations for the satellites. Not shown is a structure that would be needed to stack the satellites for dual-launch. It is anticipated that the main longerons of the lower satellite structure would be lengthened and stiffened to support the upper satellite. This approach was selected in lieu of a dual-launch adapter approach since this approach would limit the stowed diameter of the lower satellite to 4 m, and the dual-launch adapter is currently available only on the Delta IV heavy. All launch configurations are shown within a 5-m Delta IV fairing.

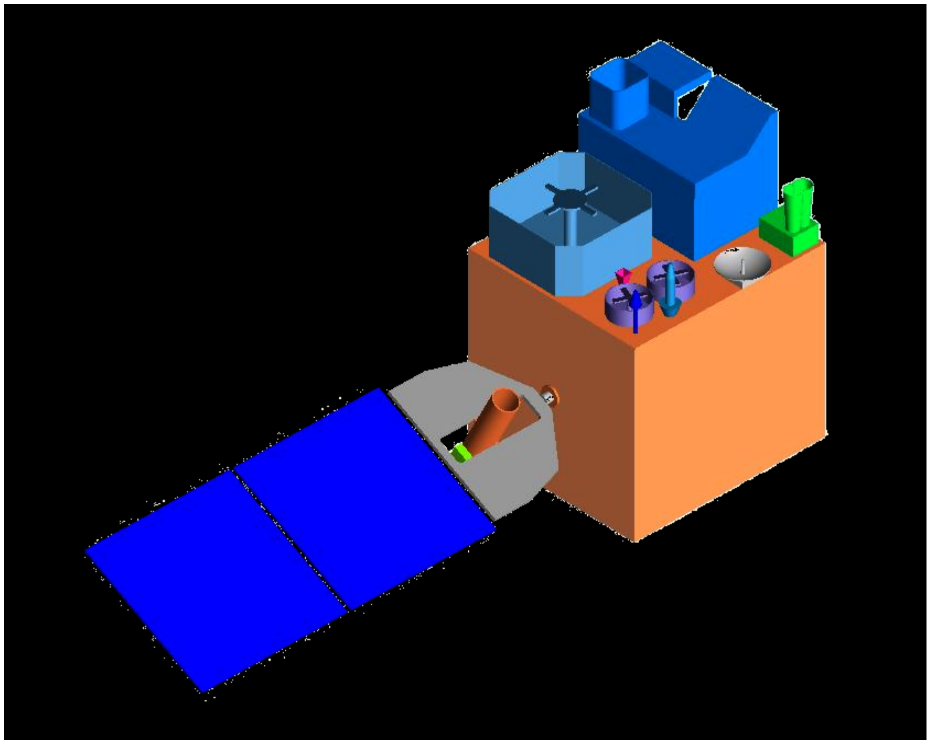

Figure 4.1. A Sat configuration.

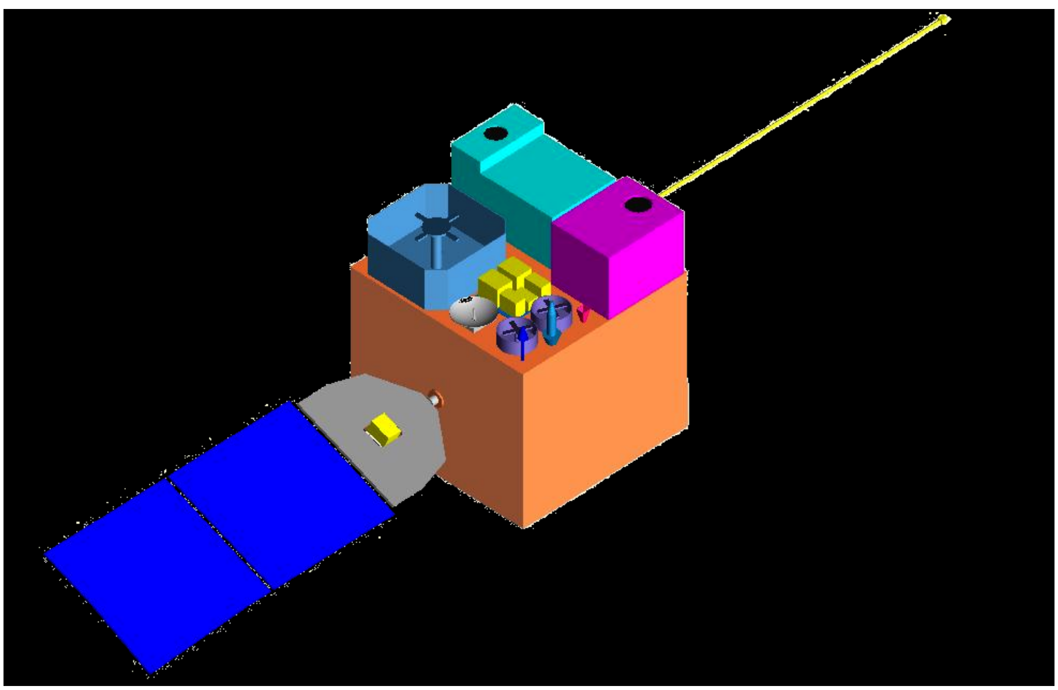

Figure 4.2. B Sat configuration.

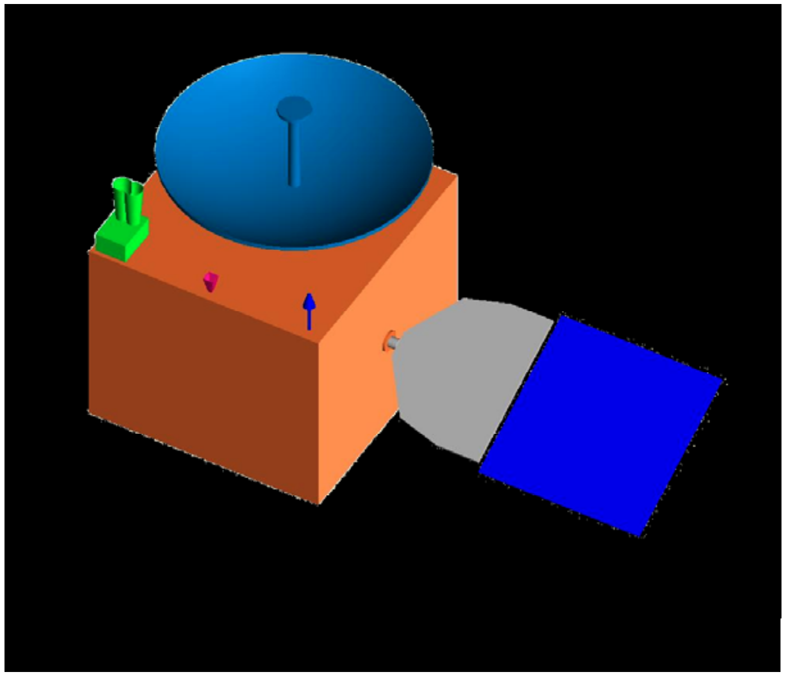

Figure 4.3. C Sat configuration.

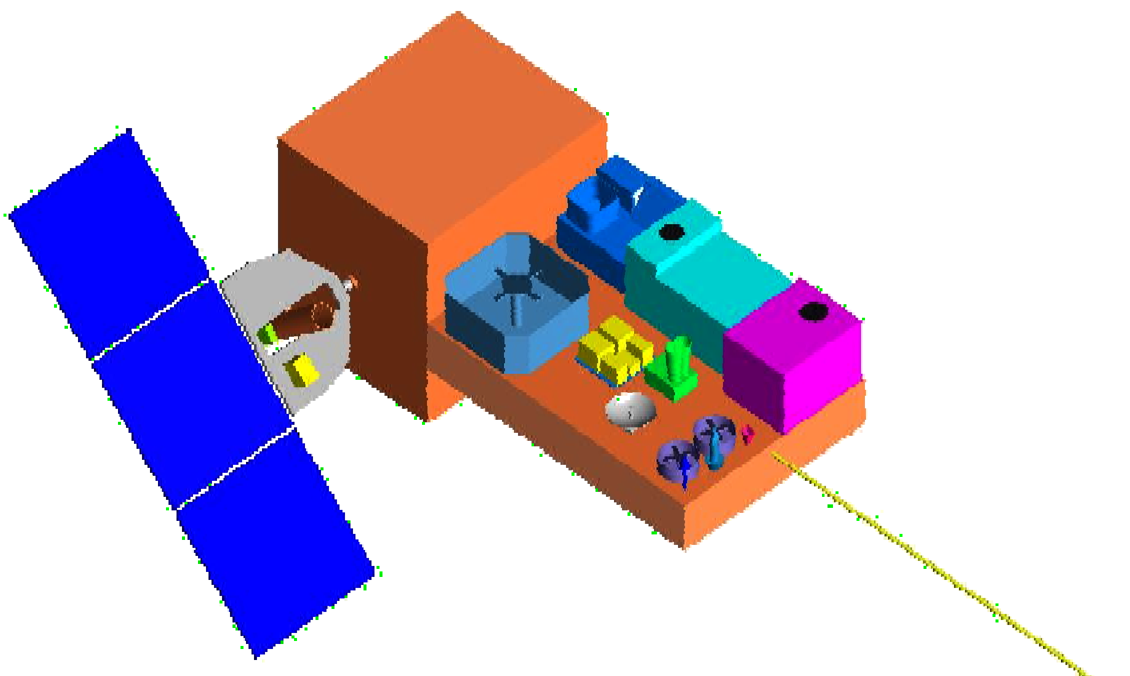

Figure 4.4. AB Sat configuration.

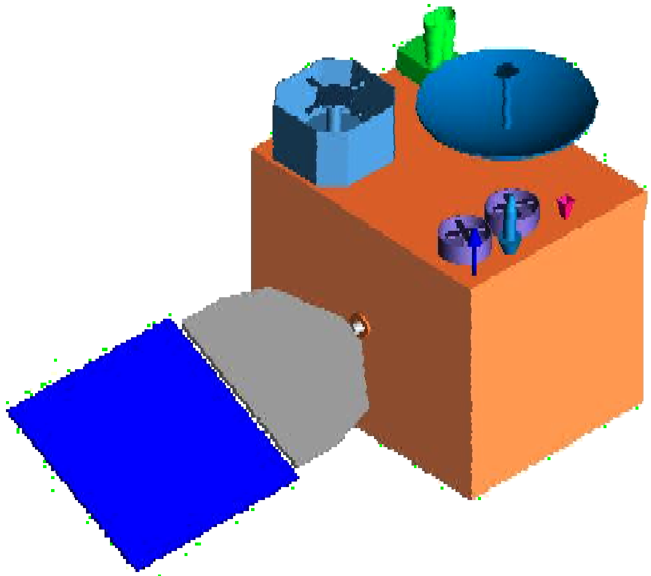

Figure 4.5. MEO Sat configuration.

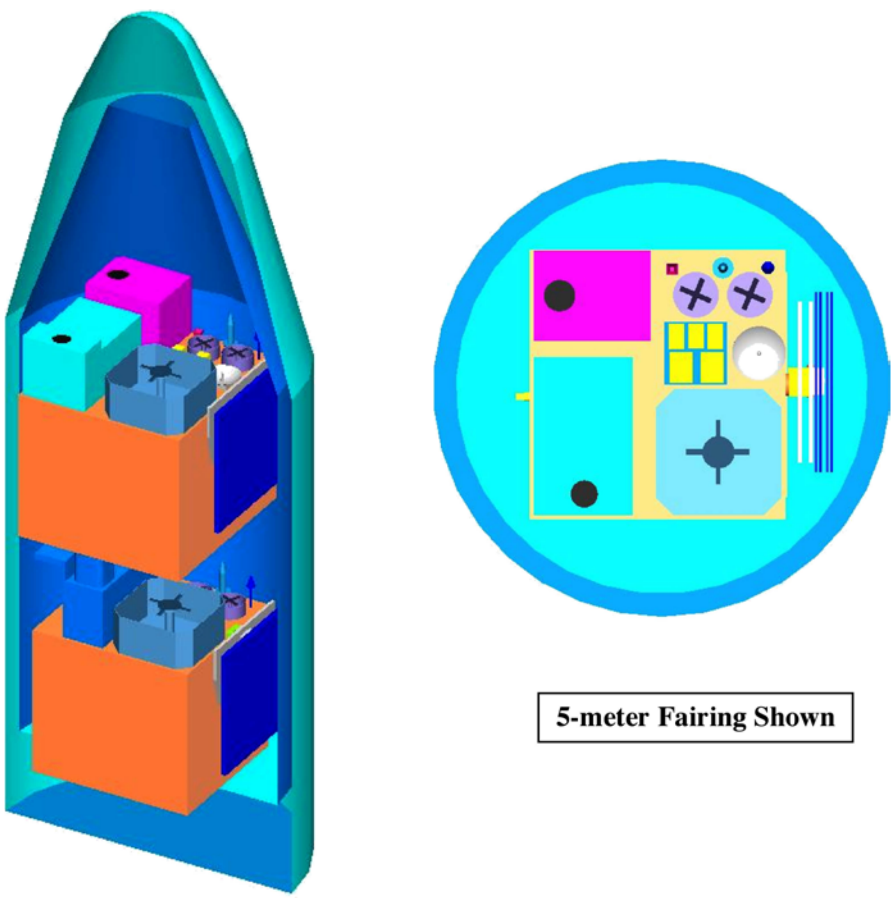

Figure 4.6. A Sat and B Sat stowed for launch.

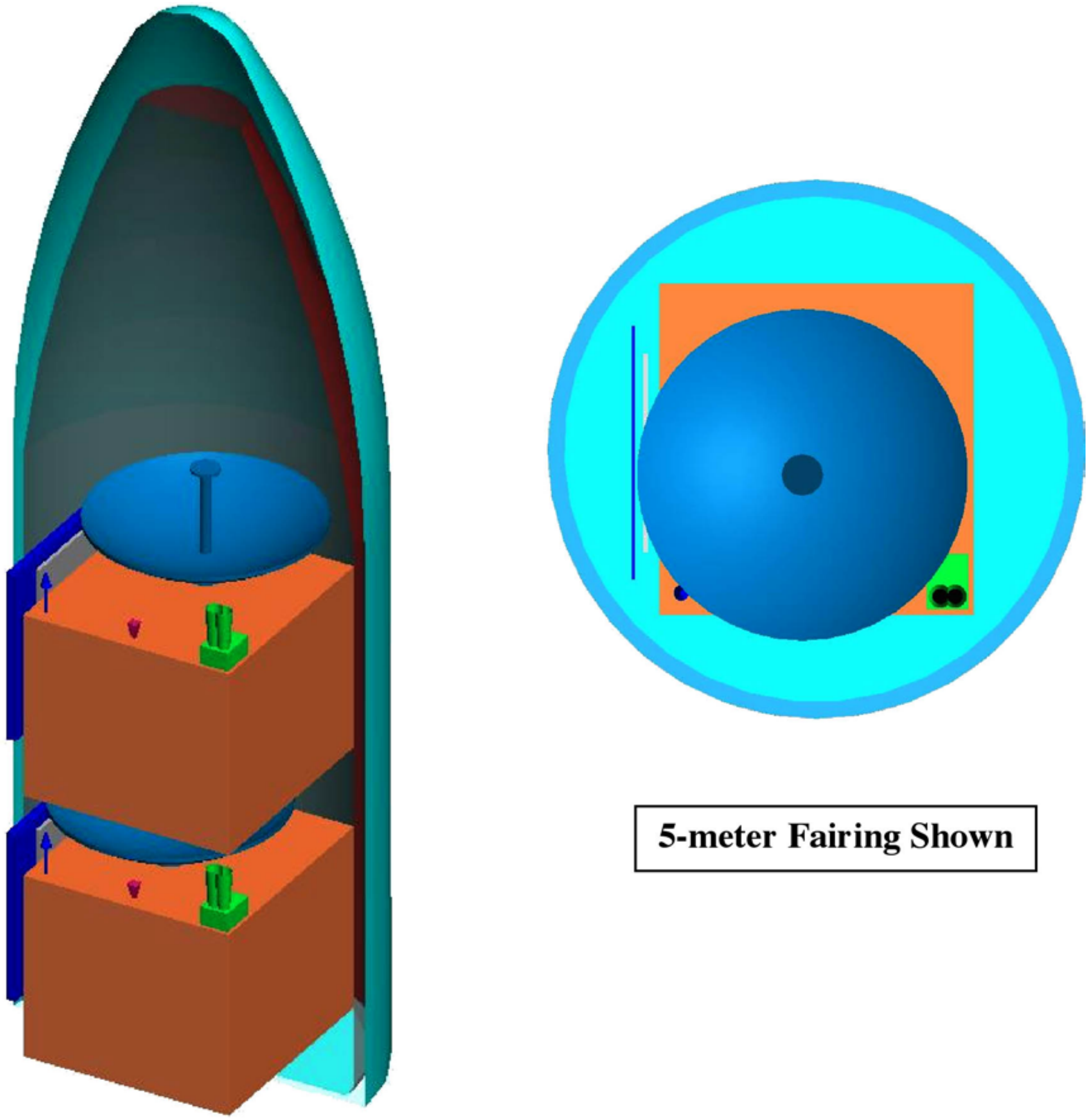

Figure 4.7. Two C Sats stowed for launch.

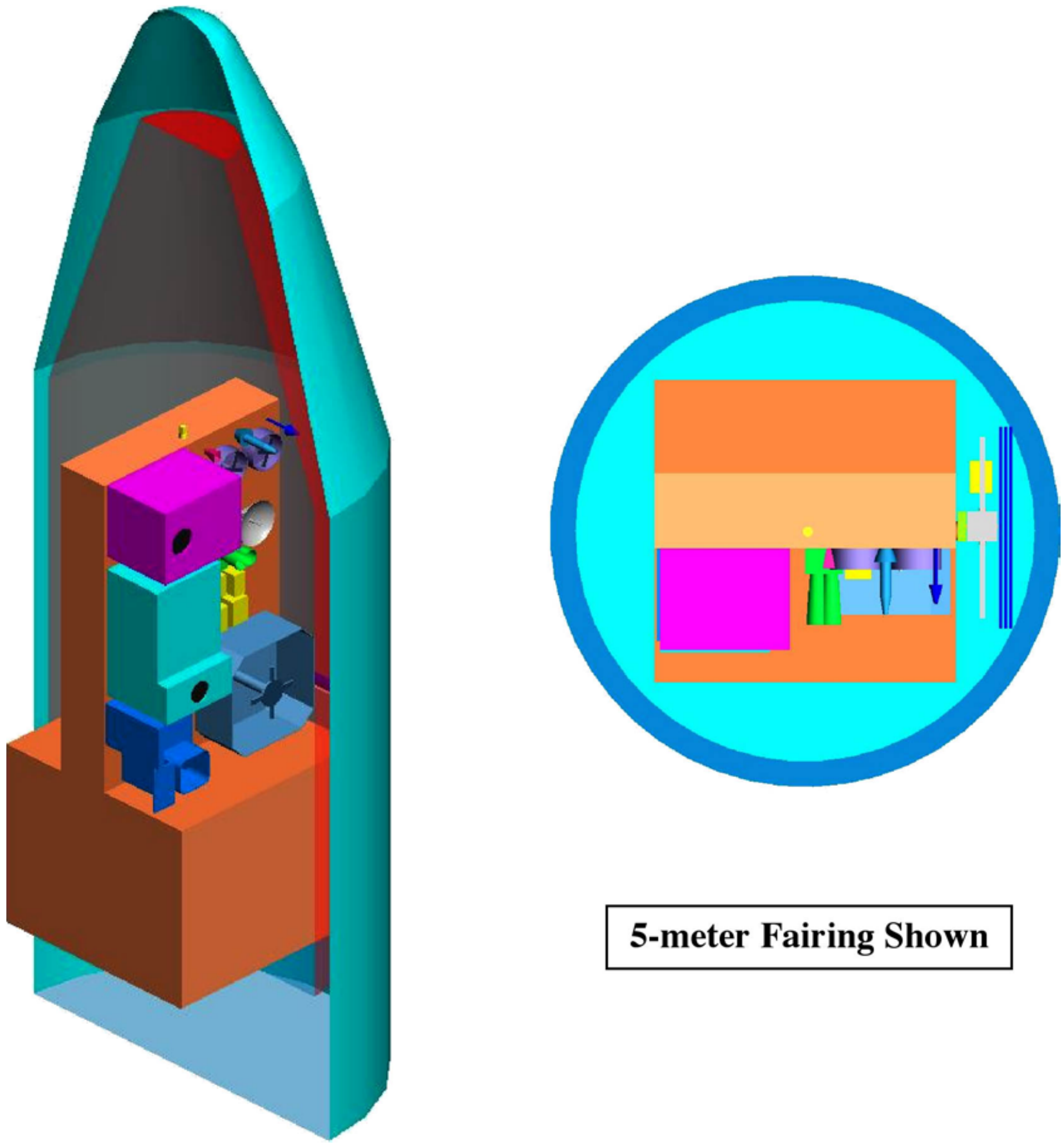

Figure 4.8. AB Sat stowed for launch.

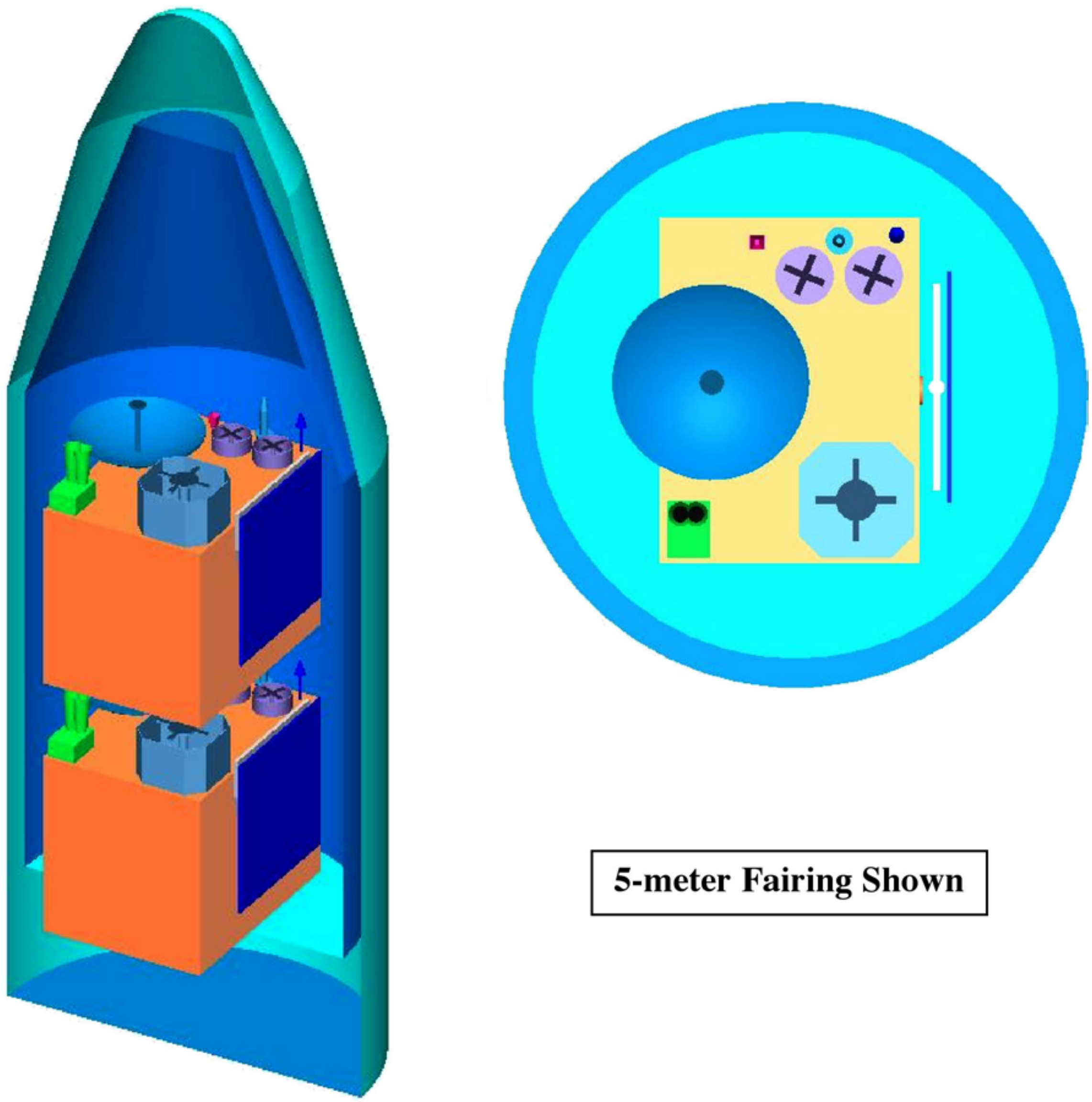

Figure 4.9. Two MEO Sats stowed for launch.

5. Astrodynamics

Tom Lang and Laura Speckman

5.1 Overview

For this study, three different architectures were explored: (1) the ABC architecture, (2) the consolidated spacecraft architecture, and (3) the MEO satellite architecture

5.2 ABC Architecture

In this architecture, the total constellation consists of 6 satellites, with one set of 3 satellites in 75°W and the other set of 3 satellites in 137°W. The spacecraft will be launched into a GEO transfer orbit and then moved to a checkout orbit at 95°W. The spacecraft are then moved to a 105 W storage orbit before being moved to either the 75°W or 137°W location.

North/South and East/West stationkeeping were sized for 9 years since the spacecraft is stored for 1 year on the ground. To hold a GEO satellite at ±0.5° inclination requires 50 m/s per year, so N/S stationkeeping delta-V is 50 m/s per year x 9 years = 450 m/s. East/West stationkeeping for GEO is 2 m/s per year, so E/W stationkeeping is 2 m/s x 9 years = 18 m/s. There is no drag makeup delta-V at GEO.

A total of 8 repositions per satellite were requested by the customer: 6 at 1°/day and 2 at 3°/day, for a total delta-V of 68.2 m/s. At satellite end-of-life, disposal to an orbit +300 km above GEO requires a delta-V of 10.9 m/s.

The traditional arrangement of 3 satellites in a single GEO slot is to place them equally spaced in longitude within a box of a certain extent. In this case, three satellites can be separated by 0.5° within the 1° box in longitude. The satellites may also be placed in a "halo" arrangement, separated by 0.866° within a 1° box in inclination and longitude; stationkeeping delta-V would be increased by approximately 10% for a halo arrangement.

5.3 AB Sat Architecture

In this architecture, the GEO satellites "A" and "B" from the ABC architecture are combined into a single spacecraft. The total constellation consists of 4 satellites, with 2 satellites in 75°W and 2 satellites in 137 W, excluding any C Sats that may be inserted. All stationkeeping, repositioning, and disposal delta-V values are the same for the consolidated spacecraft architecture case as for the ABC architecture case.

5.4 MEO Satellite Architecture

In this architecture, three MEO satellites were evenly spaced in an equatorial orbit (Walker 3/1/0 constellation) at an altitude of 19,000 km. North/South stationkeeping delta-V was 0 m/s for the MEO architecture. This is typical for MEO constellations since there is usually no North/South stationkeeping requirement for MEO spacecraft. East/West stationkeeping is 1 m/s per year, so East/West delta-V is 1 m/s x 15 years = 15 m/s. There is no drag makeup delta-V at MEO. A total of 3 repositions at 3° per day required a total of 33.3 m/s of maneuver delta-V. At satellite end-of-life, disposal to +781 km above MEO required a delta-v of 59.6 m/s.

6. Command and Data Handling

Douglas Daughaday and Ron Selden

6.1 Overview

The Command and Data Handling (C&DH) subsystem is fully redundant due to the length of the mission lifetime. The C&DH subsystem consists of a processor, two input/output controller boards, two remote interface units, a solid-state mass memory device, a chassis, and a power supply. All of the components are redundant except for the mass storage, which is internally redundant. Figure 6.1 presents a block diagram of the C&DH model. The C&DH architecture did not change between the configurations in this study.

Table 6.1 presents the details of the C&DH subsystem design and is the same for all configurations considered for this study.

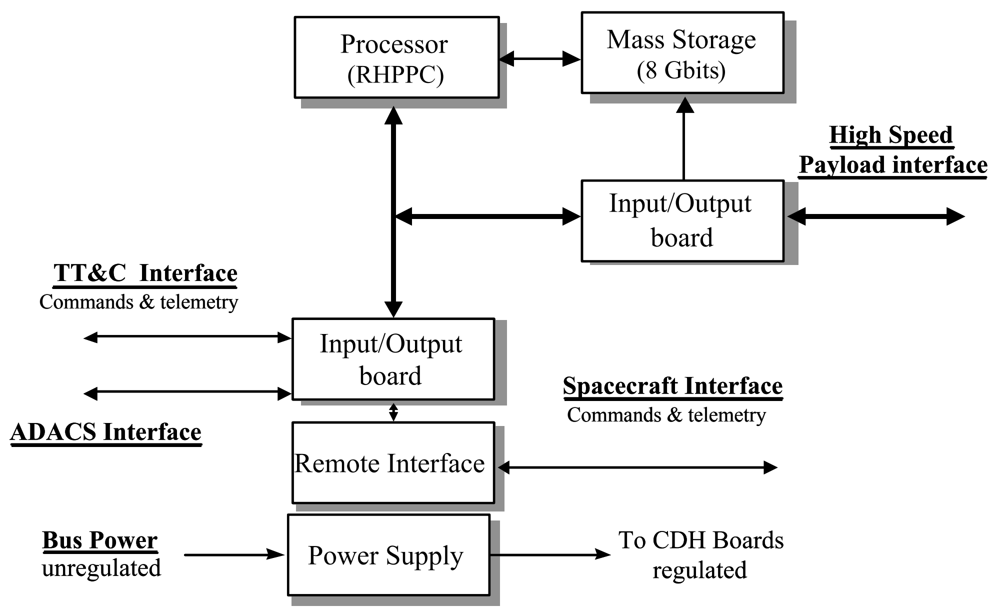

Figure 6.1. C&DH model block diagram.

Table 6.1. C&DH Subsystem Design Summary

	Units	Mass (kg)	Power (W)	NASA TRL
Command & Data Handling		**11.2**	**15.3**	**6**
Processor	2	1.2	6.0	7
Input / Output Controller	2	1.0	2.5	5
Solid State Memory	2	5.0	3.0	6
Power supply	2	2.0	3.8	7
Chassis	1	2.0	0.0	7

6.2 Technology Assumptions

There are no special technology developments taken into account for this study. The power and mass numbers are based on current technology, and no significant improvements are foreseen by the technology freeze date of 2008.

The high-speed input/output board that interfaces to the payload equipment is a new design and may require significant development efforts. The high-speed payload interface is envisioned to be a Firewire or equivalent connection. A detailed trade study must be completed to determine the specific hardware design and approach for this interface.

The capacity of the mass memory is a high estimate. An in-depth look at the required capacity needs to be completed. The estimate used for this study should be considered as an upper bound for the capacity of the mass memory.

6.3 Component Descriptions

Processor**:** Rad-Hard Power PC (RHPPC)

- Throughput: ~200 MIPS
- Mass: 1.2 kg
- Power: 12 W

Input/output controller: Custom design

- Mass: 1.0 kg
- Power: 5 W

Input/output controller (high-speed payload interface)

- Mass: 1.0 kg
- Power: 10.0 W

Solid-state memory

- Capacity: 1 Gbits
- Mass: 5 kg
- Power: 3 W

Power supply: Estimate

- Mass: 0.5 kg
- Efficiency: 75%

Chassis:

- Mass: 2.0 kg

7. Telemetry, Tracking, and Command

John O'Donnell

The TT&C subsystem is configured to produce a design of the Command and Control rf link in support of the TT&C subsystem on the GOES Block 5 spacecraft. Commands originate from a CONUS-based control facility and are transmitted to the spacecraft by a network of remote ground facilities. Vehicle telemetry is received at the control facility by the network's return link. The TT&C subsystem is designed and sized to support the standard NASA Unified S-band link for activities that include launch, early orbit checkout, orbit transfer (if applicable), and operational orbit ranging/health status and anomaly resolution. The TT&C subsystem design assumptions are presented in Table 7.1.

The TT&C subsystem unit design is presented in Table 7.2. The mass and power estimates represent totals for the unit quantities, which provide a fully redundant system.

The TT&C subsystem is flight proven, thus the high TRL numbers in Table 7.2. All configurations studied for the GOES Block 5 spacecraft resulted in the same TT&C subsystem design, thus Table 7.2 represents all configurations.

Table 7.1. Design Assumptions per Module/Link Capability

	TT&C Uplink	**TT&C Downlink**
Frequencies	Unified S-band Command Link Frequency Band	Unified S-band Telemetry Link (2200 to 2300 MHz range)
Data Rates	Cmd = 2 kbps	Tlm = 8 kbps during launch, early orbit
		Tlm = 32 kbps during nominal on-station operations
Antennas	RTS ant = minimum 10 m dia.	Spacecraft: - Omni (qty = 2)
		- EC Horn (qty = 1)

Table 7.2. TT&C Subsystem Design

	Unit Qty	**Total Mass (lb)**	**Total Power (W)**	**NASA TRL**
TT&C Subsystem (SGLS Link)				
Omni Antenna	2	0.8	0	9
Horn Antenna Assembly	1	5.0	0	9
Transponder	2	14	32	8
Cmd Signal Conditioning Unit	2	3.6	1.5	8
TLM Base Band Assembly Unit	2	3.6	1.5	8
Local Oscillator	2	3.0	2	8
Comsec	2	2.0	2.5	9
Miscellaneous RF Hardware	1	5.0	0	9
TT&C Subsystem Total:		**37**	**39.5**	

The downlink telemetry rates of 8 kbps and 32 kbps are typical, non-stressing rates for a flight telemetry system. It is safe to anticipate that higher rate vehicle/payload telemetry could be transmitted "in-band" in the payload sensor data downlink to the Wallops ground site from the mission orbit. The Unified S-band link through the Ground Control Network would be available for scheduled ranging and states of health contacts and anomaly resolution contacts. During launch and orbit transfer the TT&C system would operate through the two hemispherical coverage patch antennas providing the vehicle near-2π sr coverage. During this time, The anticipated data rate is 8 kbps.

Once on orbit at GEO, the TT&C system would switch to an Earth coverage horn antenna providing sufficient gain for increasing the telemetry rate to 32 kbps. The link requires a 10-W rf SSPA transmitter for link closure during all mission orbit phases. At the time of GOES Block 5 satellite development, a SGLS-USB dual-mode transponder will be available off the shelf with ranging turnaround, thus allowing for command and control compatibility between NASA and Air Force Satellite Control Network ground stations. The Command Signal Conditioning Unit (CSCU) provides the command decoder function and is the forward link interface to the C&DH subsystem. The Telemetry Baseband Assembly Unit (BBAU) provides the associated functional interface between the C&DH subsystem and telemetry return link for telemetry encoding.

Both the SCU and the BBAU can be integrated slices within the C&DH. The comsec unit is anticipated to be the L3Comm MFU flight-qualified unit providing Cardholder command decryption/ authentication and Pegasus telemetry encryption.

8. Payload Communications

John O'Donnell

8.1 Overview

For each of the configurations studied, the GOES Block 5 Payload Communication model is designed to satisfy the data requirements of the payloads as proposed by the customer and provide a means of evaluating the overall delta impact to the spacecraft design. Although this summary write-up provides an allocation of mass and powers at the individual communication subsystem levels, the actual payload communication model used during the study evaluated the communication design at a unit level mass and power breakdown, and these details could be made available. During the CDC session, many spacecraft configurations were examined. This report summarizes the primary five configurations and two excursions.

The payload communications subsystem provides the capability to transmit raw sensor data directly to the ground station located at Wallops Island, receive the processed mission data uplink from Wallops Island, and transpond that uplink via a broadcast mode to all in-view ground users. The concept behind this study was to have up to three spacecraft occupying common orbital slots at 75 West and 137 West. Each spacecraft would have a raw sensor data downlink transmission to Wallops Island; yet only one spacecraft (spacecraft A or B) would actively provide the global re-broadcast of processed mission data. Before examining the design of the payload communications subsystem for each of the study configurations, an overview of the communication links and their options is provided.

Sensor Data Downlink. This communication link is a point-to-point rf link from each of the GOES Block 5 satellites to the Wallops Island Ground facility. It provides raw sensor data collected by the satellite for processing on the ground. A single frequency band was examined for this link: X-band (8215–8400 MHz). Based on the available spectrum and the required downlink data rates, all three spacecraft would frequency share the X-band spectrum. The modulation of this data link was assumed to be O-QPSK with $15/16^{th}$ Turbo code forward error correction (FEC) applied.

Global Rebroadcast (GRB). This communication link provides processed data to the user community. It is a transponded link from Wallops Island (uplink) through the GOES Block 5 satellite and globally broadcast to the user community (downlink). A single frequency band combination was examined for this transponded link: X-band uplink (7190–7235 MHz) and L-band downlink broadcast (1683–1695 MHz). For compatibility with the existing GOES GRB transmission, this link employed O-QPSK modulation and no forward error correction. As mentioned, only one of the three spacecraft in each orbital slot would provide this global rebroadcast.

Auxiliary Signal Broadcast. This communication link provides three low-rate, frequency-multiplexed auxiliary data transmissions to the user community. These multiplexed signals are transmitted from Wallops Island and are transponded by the satellite for user broadcast reception. All configurations occupied the uplink/downlink spectrum of 7190–7235 MHz/ 1695–1698 MHz. The three signal set consists of:

- LRIT: 600 kbps with QPSK modulation
- EMWIN: 8 kbps with BPSK modulation
- DCPR: 233 channels of 100, 300, and 1200 bps

The spacecraft providing the GRB transmission would also simultaneously provide this auxiliary signal broadcast transmission.

As mentioned, this report will summarize the five primary study configurations resulting from the CDC sessions. Each of these configurations will be discussed in order. All of the design information to be presented represents the design of the communication system without mass and power contingency included. Within the Systems module of the Space Systems CDC, a 25% contingency is added to the communication system's mass and power estimates. The spacecraft bus is then sized based on these estimates that include contingency.

8.2 A Sat

This configuration involved the following communication system design characteristics:

Sensor Data Downlink: X-band downlink at 58 Mbps. This direct downlink to Wallops was designed as a single polarization transmission using O-QPSK modulation with 15/16 FEC. Detailed link analysis defined a transmit power of 7.0 W rf (linear) and a 0.5-m gimbaled antenna to provide 4 dB of link margin to the Wallops receive system. The linear operation of the X-band SSPA was defined at 2 dB back-off from saturation. The link margin was increased to 4 dB from the baseline of 3 dB to account for anticipated losses in the X-band diplexer implemented within the space-ground communication system.

GRB: X-band/L-band uplink/broadcast at 5 Mbps. The uplink signal is received through the same 0.5-m gimbal antenna used for the sensor data downlink to Wallops. The link analysis showed that the available downlink broadcast spectrum of 1683 to 1695 MHz required O-QPSK modulation and 24 W rf (linear) power through a planar cup dipole antenna providing 15.3 dBi of gain at 5° elevation. Previous analysis had indicated that 4.1 dB of back-off from saturation was required, when using O-QPSK modulation, for operation of the TWTA.

Auxiliary: X-band/L-band uplink/broadcast. As described, this signal set is transponded by the GOES Block 5 satellite to the user community. Link analysis showed that the LRIT signal required 10 W rf (linear), the EMWIN signal required 5 W rf (linear), and the DCPR signal required 8 W rf (linear) through a planar cup dipole antenna providing 15.3 dBi of gain at 5° elevation. In all cases, linear operation of the SSPAs was at a 2 dB back-off point.

A conceptual overview of this communication configuration is provided in Figure 8.1.

Table 8.1 summarizes in a top-level breakdown the mass and power of the communication subsystem.

As noted, the L-band global broadcast required 24 W of rf power operating in the linear region. For operation of the TWTA, recent analysis indicates that a level of 4.1 dB of back-off is required for O-QPSK modulation. For TWTA RF to DC conversion, a 48% of tube efficiency was assumed.

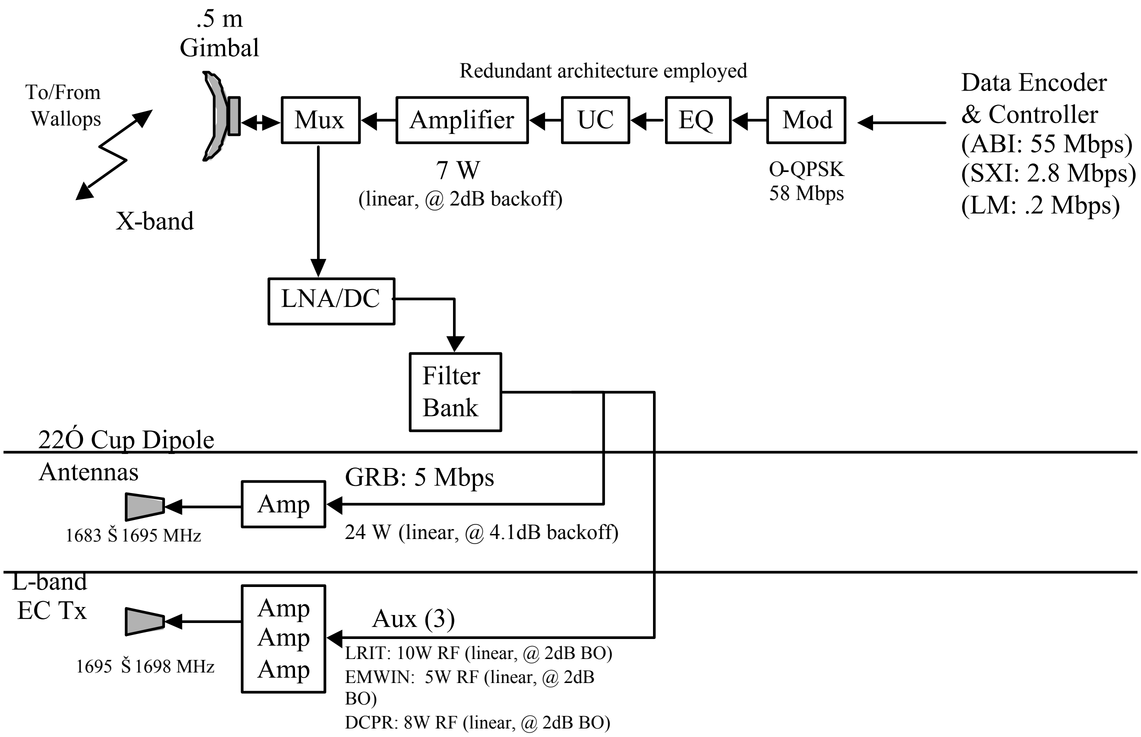

Figure 8.1. Spacecraft A communication system block diagram.

Table 8.1. A Sat Communication Summary

A Sat	Note	Mass (lb)	Power (W)
Wallops Antenna System	0.5 m	31.4	6.0
Broadcast Antenna Systems	Planar Cup Dipoles	17.6	0.0
Sensor Data Downlink rf Hardware	X-band	16.7	44.8
Sensor Data Downlink Electronics Hardware	Single pol O-QPSK	32.4	21.0
Broadcast System rf Hardware (GRB & Aux)	L-band O-QPSK (GRB)	54.6	267.8
	Communication System Total:	**153**	**340**

8.3 A Sat without Global Broadcast Capability

This configuration is an excursion from the “baseline” A Sat. It examined the impact of removing the communication hardware associated with the broadcast for GRB and Auxiliary signals. A conceptual overview of this communication configuration is provided in Figure 8.2.

Table 8.2 summarizes in a top-level breakdown the mass and power of the communication subsystem.

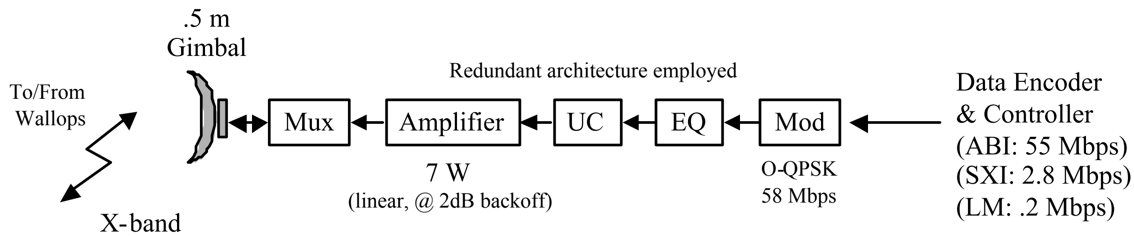

Figure 8.2. A Sat excursion communication system block diagram.

Table 8.2. A Sat Excursion Communication Summary

A Sat Excursion	Note	Mass (lb)	Power (W)
Wallops Antenna System	0.5 m	31.4	6.0
Broadcast Antenna Systems		0.0	0.0
Sensor Data Downlink rf Hardware	X-band	16.7	44.8
Sensor Data Downlink Electronics Hardware	Single pol O-QPSK	32.4	21.0
Broadcast System rf Hardware (GRB & Aux)		0.0	0.0
Communication System Total:		**81**	**72**

8.4 B Sat

This configuration involved the following communication system design characteristics:

Sensor Data Downlink. X-band downlink at 67 Mbps. This direct downlink to Wallops was designed as a single polarization transmission using O-QPSK modulation with 15/16 FEC. Detailed link analysis defined a transmit power of 8.0 W rf (linear) and a 0.5-m gimbaled antenna to provide 4 dB of link margin to the Wallops receive system. The linear operation of the X-band SSPA was defined at 2 dB back-off from saturation. The link margin was increased to 4 dB from the baseline of 3 dB to account for anticipated losses in the X-band diplexer implemented within the space-ground communication system.

GRB. X-band/L-band uplink/broadcast at 5 Mbps. The uplink signal is received through the same 0.5-m gimbal antenna used for the sensor data downlink to Wallops. The link analysis showed that the available downlink broadcast spectrum of 1683 to 1695 MHz required O-QPSK modulation and 24 W rf (linear) power through a planar cup dipole antenna providing 15.3 dBi of gain at 5° elevation. Previous analysis had indicated that 4.1 dB of back-off from saturation was required, when using O-QPSK modulation, for operation of the TWTA. The design of this broadcast system is the same as the A Sat design.

Auxiliary. X-band/L-band uplink/broadcast. As described, this signal set is transponded by the GOES Block 5 satellite to the user community. Link analysis showed that the LRIT signal required 10 W rf (linear), the EMWIN signal required 5 W rf (linear), and the DCPR signal required 8 W rf (linear) through a planar cup dipole antenna providing 15.3 dBi of gain at 5° elevation. In all cases, linear operation of the SSPAs was at a 2 dB back-off point. The design of this broadcast system is the same as the A Sat design.

A conceptual overview of this communication configuration is provided in Figure 8.3.

Table 8.3 summarizes in a top-level breakdown the mass and power of the communication subsystem.

As mentioned, either the A Sat or B Sat spacecraft would have an active GRB and Aux broadcast transmission. The payload communication subsystem includes the associated dc power for this capability to ensure that both spacecraft buses are sized to handle the worst-case mass and power load.

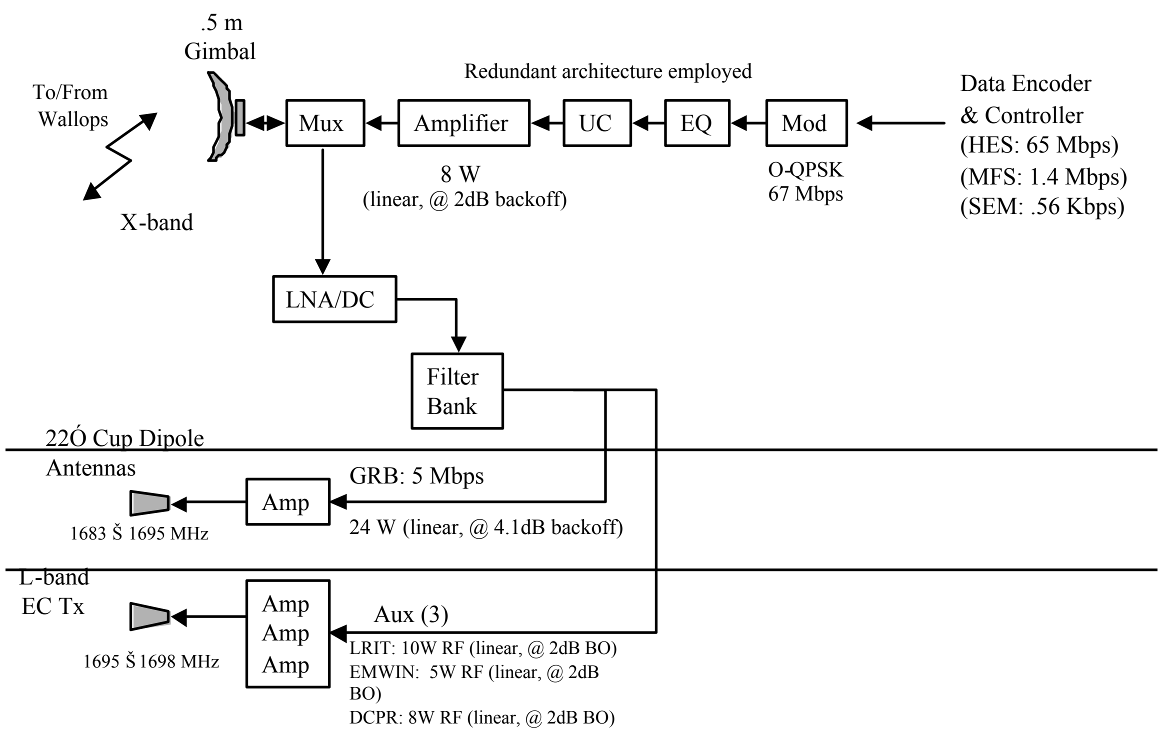

Figure 8.3. B Sat communication system block diagram.

Table 8.3. B Sat Communication Summary

B Sat	Note	Mass (lb)	Power (W)
Wallops Antenna System	0.5 m	31.4	6.0
Broadcast Antenna Systems	Planar Cup Dipoles	17.6	0.0
Sensor Data Downlink rf Hardware	X-band	16.7	49.5
Sensor Data Downlink Electronics Hardware	Single pol O-QPSK	34.7	22.1
Broadcast System rf Hardware (GRB & Aux)	L-band O-QPSK (GRB)	54.6	267.8
	Communication System Total:	**155.0**	**345.3**

8.5 B Sat without Global Broadcast Capability

This spacecraft is an excursion to the B Sat spacecraft. It examined the impact of removing the communication hardware associated with the broadcast for GRB and Auxiliary signals. A conceptual overview of this communication configuration is provided in Figure 8.4.

Table 8.4 summarizes in a top-level breakdown the mass and power of the communication subsystem.

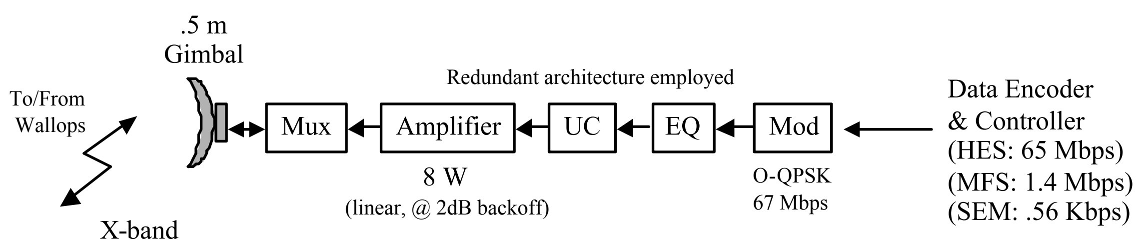

Figure 8.4. B Sat excursion communication system block diagram.

Table 8.4. B Sat Excursion Communication Summary

B Sat Excursion	Note	Mass (lb)	Power (W)
Wallops Antenna System	0.5 m	31.4	6.0
Broadcast Antenna Systems		0.0	0.0
Sensor Data Downlink rf Hardware	X-band	16.7	49.5
Sensor Data Downlink Electronics Hardware	Single pol O-QPSK	34.7	22.1
Broadcast System rf Hardware (GRB & Aux)		0.0	0.0
	Communication System Total:	**83**	**78**

8.6 C Sat

This configuration involved the following communication system design characteristics:

Sensor Data Downlink. X-band downlink at 2.1 Mbps. This direct downlink to Wallops was designed as a single polarization transmission using O-QPSK modulation with 15/16 FEC. Detailed link analysis defined a transmit power of 5.2 W rf (linear) and an Earth coverage horn antenna to provide 3 dB of link margin to the Wallops receive system. The linear operation of the X-band SSPA was defined at 2 dB back-off from saturation.

GRB. C Sat does not have a global rebroadcast capability.

Auxiliary. C Sat does not have an auxiliary signal broadcast capability.

A conceptual overview of this communication configuration is provided in Figure 8.5.

Table 8.5 summarizes in a top-level breakdown the mass and power of the communication subsystem.

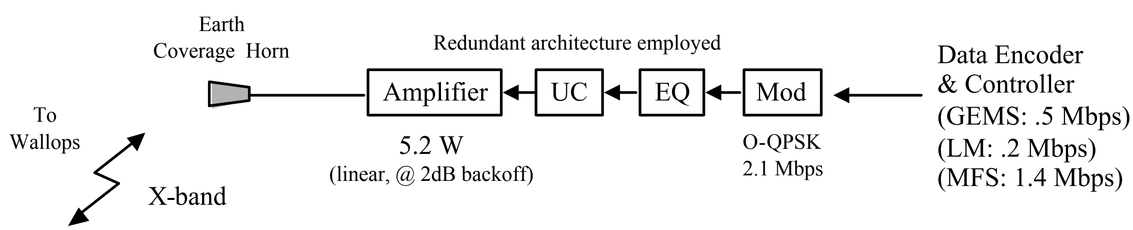

Figure 8.5. C Sat communication system block diagram.

Table 8.5. C Sat Communication Summary

C Sat	Note	Mass (lb)	Power (W)
Wallops Antenna System	EC Horn	7.7	0.0
Broadcast Antenna Systems		0.0	0.0
Sensor Data Downlink rf Hardware	X-band	13.9	33.3
Sensor Data Downlink Electronics Hardware	Single pol O-QPSK	14.3	9.5
Broadcast System rf Hardware (GRB & Aux)		0.0	0.0
Communication System Total:		**36.0**	**42.8**

8.7 AB Sat

This configuration involved the following communication system design characteristics:

Sensor Data Downlink. X-band downlink at 91 Mbps. This direct downlink to Wallops was designed as a single polarization transmission using O-QPSK modulation with 15/16 FEC. Detailed link analysis defined a transmit power of 11 W rf (linear) and a 0.5-m gimbaled antenna to provide 4 dB of link margin to the Wallops receive system. The linear operation of the X-band SSPA was defined at 2 dB back-off from saturation. The link margin was increased to 4 dB from the baseline of 3 dB to account for anticipated losses in the X-band diplexer implemented within the space-ground communication system.

GRB. X-band/L-band uplink/broadcast at 5 Mbps. The uplink signal is received through the same 0.5-m gimbal antenna used for the sensor data downlink to Wallops. The link analysis showed that the available downlink broadcast spectrum of 1683 to 1695 MHz required O-QPSK modulation and 24 W rf (linear) power through a planar cup dipole antenna providing 15.3 dBi of gain at 5° elevation. Previous analysis had indicated that 4.1 dB of back-off from saturation was required, when using O-QPSK modulation, for operation of the TWTA.

Auxiliary. X-band/L-band uplink/broadcast. As described, this signal set is transponded by the GOES Block 5 satellite to the user community. Link analysis showed that the LRIT signal required 10 W rf (linear), the EMWIN signal required 5 W rf (linear), and the DCPR signal required 8 W rf (linear) through a planar cup dipole antenna providing 15.3 dBi of gain at 5° elevation. In all cases, linear operation of the SSPAs was at a 2dB back-off point.

A conceptual overview of this communication configuration is provided in Figure 8.6

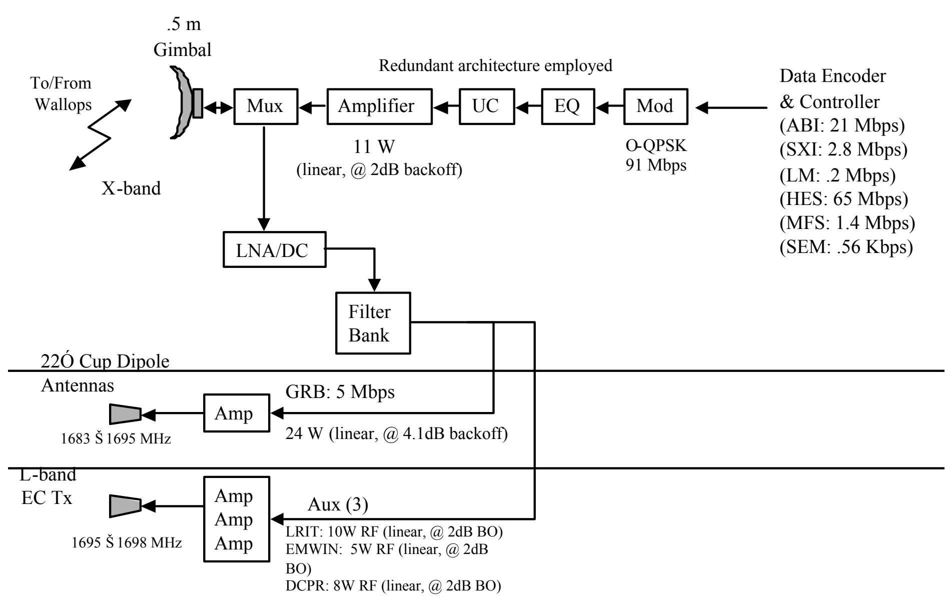

Figure 8.6. AB Sat communication system block diagram.

Table 8.6 summarizes in a top-level breakdown the mass and power of the communication subsystem.

Table 8.6. AB Sat Communication Summary

AB Sat	Note	Mass (lb)	Power (W)
Wallops Antenna System	0.5 m	31.4	6.0
Broadcast Antenna Systems	Planar Cup Dipoles	17.6	0.0
Sensor Data Downlink rf Hardware	X-band	19.7	65.6
Sensor Data Downlink Electronics Hardware	Single pol O-QPSK	39.3	25.2
Broadcast System rf Hardware (GRB & Aux)	L-band O-QPSK (GRB)	54.6	267.8
Communication System Total:		**162.6**	**364.6**

8.8 MEO Sat

The concept of the communication system on the MEO spacecraft is to provide global, as opposed to hemispherical, broadcast of the processed sensor data and the auxiliary signals among other trades. The communication system includes the following characteristics:

GRB. X-band/L-band uplink/broadcast at 5 Mbps. The uplink signal (multiplexed GRB and Auxiliary signals transmitted from Wallops) is received through a gimbal horn antenna. The link analysis for the MEO orbit showed that the available downlink broadcast spectrum of 1683 to 1695 MHz required O-QPSK modulation and 21 W rf (linear) power through a planar cup dipole antenna providing 11.3 dBi of gain at 5° elevation. The broadcast antennas are smaller in size compared to the GEO application in order to provide the same ground coverage as obtained from GEO. As such, their associated gain is less than the GEO versions. But this delta in performance is nearly offset by the delta in path loss of a GEO spacecraft versus a MEO spacecraft. Previous analysis had indicated that 4.1 dB of back-off from saturation was required, when using O-QPSK modulation, for operation of the TWTA. The design of this broadcast system is the same as the B Sat design.

Auxiliary. X-band/L-band uplink/broadcast. As described, this signal set is transponded by the GOES Block 5 satellite to the user community. The link analysis for the MEO orbit showed that the LRIT signal required 9 W rf (linear), the EMWIN signal required 4.3 W rf (linear), and the DCPR signal required 7 W rf (linear) through a planar cup dipole antenna providing 11.3 dBi of gain at 5° elevation. In all cases, linear operation of the SSPAs was at a 2 dB back-off point. The design of this broadcast system is the same as the B Sat design. A conceptual overview of this communication configuration is provided in Figure 8.7

Table 8.7 summarizes in a top-level breakdown the mass and power of the communication subsystem.

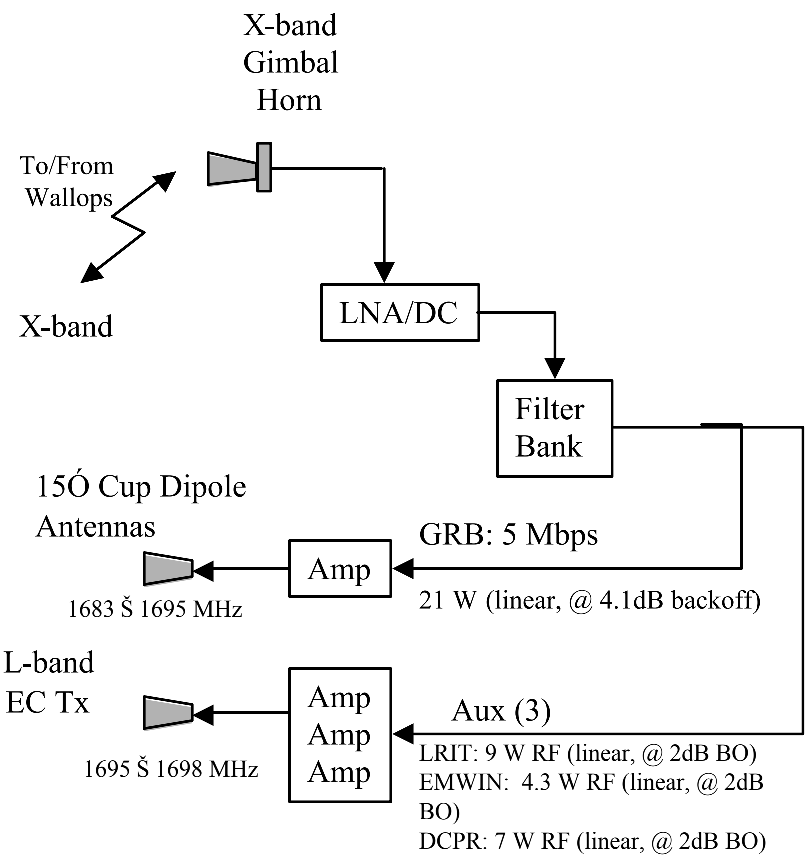

Figure 8.7. MEO Sat communication system block diagram.

Table 8.7. MEO Sat Communication Summary

MEO Sat	Note	Mass (lb)	Power (W)
Wallops Antenna System	Gimbal Horn	15.2	4.0
Broadcast Antenna Systems	Planar Cup Dipoles	15.4	0.0
Uplink rf Hardware	X-band	8.3	2.1
Uplink Electronics Hardware		8.9	3.2
Broadcast System rf Hardware (GRB & Aux)	L-band O-QPSK (GRB)	62.8	228.5
	Communication System Total:	**111**	**238**

Due to the radiation environment at the anticipated MEO orbit, additional shielding was applied to the electronics hardware supporting reception of the uplink transmission and broadcast of the processed sensor data and auxiliary data. The extended on-orbit mission life of 15 years is supported by the implementation of a fully redundant rf and electronics hardware design.

8.9 Summary

The payload communications subsystem provides either the capability to transmit raw sensor data directly to Wallops Island, or the capability to globally re-broadcast processed data and auxiliary data signals received from Wallops Island to globally distributed users, or the combination of both functions. The three spacecraft in each of the GEO orbital slots (75 West and 137 West) frequency share the X-band spectrum (8215–8400 MHz) for their direct downlink transmissions. Either A Sat or B Sat in the orbital trio provides the active rebroadcast capability, although both Spacecraft A and Spacecraft B have the capability to fulfill this mission. The design of the global rebroadcast system is the same for each spacecraft: globally transpond 5 Mbps of processed data in the L-band spectrum of 1683–1695 MHz; and globally transpond three auxiliary signals in the L-band spectrum of 1695–1698 MHz. The dc power requirements of the amplifiers for each of these links were based on operation in the linear region (2.0 dB backoff for the mission data link, 4.1 dB backoff for the GRB broadcast, and 2.0 dB backoff for broadcast of the auxiliary signals). The selection of these spectrum bands and these amplifier backoff levels is based on recent study activities currently being conducted within the Communications Systems Subdivision at The Aerospace Corporation in support of NOAA.

An issue to be noted for this three-satellite-per-orbital-slot constellation is that due to the separation of the three spacecraft in each orbital slot, three individual ground antennas are required at Wallops Island for active command and control and reception of the raw sensor data downlink. The MEO Sat require the same functional hardware for global re-broadcast as do the GEO spacecraft, the only exception is that the MEO Sat require additional shielding for the electronics hardware due to the radiation environment observed at MEO.

9. Attitude Determination and Control

Andrei Doran

9.1 Overview

The attitude control approach used was the standard three-axis stabilized type. It was the only approach that could achieve the high pointing precision needed. There were several attitude determination and control subsystem (ADACS) areas requiring attention for this mission. The most important issue was the attitude determination accuracy. The requirement, 3 arc-sec, was stringent, but does not push the envelope of satellites such as Chandra (where Ball star sensors achieved sub arc-second 3σ accuracy). Discussions during the study indicated that 7 arc-sec would also be acceptable for the attitude determination requirement. However, since it did not lead to a different sensor selection, the 3 arc-sec requirement was used.

An analysis of the combined pointing knowledge obtainable with the best star sensors and gyros showed that the 3 arc-sec specifications can be met. The analysis used the same tool introduced in the first GOES-R CDC study.* This tool calculated the combined error variance based on the individual accuracy of all the sensors. However, the tool has been refined, and subsequently, it provides more accurate values now. The sensor selection and analysis are described under the attitude determination section of this report.

A related issue was the jitter control and knowledge. There was the usual notation problem, in that the jitter specification is not stated as a function of frequency, but rather as a total specification on angular rate. The ABI drives both the jitter control specification and the knowledge requirement. The jitter control value was well within the sensing capability of the selected gyros, and achievable with normal control algorithms and actuator dynamics. However, the knowledge requirement will be tough to meet, unless the already small gyro bias can be calibrated and reduced significantly with star sensor data in the Kalman filter. A resolution of this issue requires further study, and is discussed at the end of this section.

Another issue was the single solar panel design. While convenient mechanically (fewer gimbals, cables, etc.), it lead to a more imbalanced structure, with a larger center-of-mass (CM) to center-of-pressure (CP) distance, and incurs a larger solar pressure torque than would a two solar panel design. The issue was not overwhelming, and did not drive the reaction wheel (RW) size since the accumulated momentum from solar pressure torques over an orbit was less than the momentum needed for the slew maneuver. The slew requirement, a 180° yaw flip twice a year for thermal and solar panel sun viewing reasons, generated approximately twice the momentum accumulation generated by the solar pressure during an orbit. The cyclic nature of the solar torque for a nadir-pointing satellite in GEO helps. The RWs had to handle the accumulation of half an orbit since the next half cancels out the first. Only a small part of the solar torque is cumulative, so the amount of fuel needed to unload

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* National Oceanic and Atmospheric Administration Geostationary Operational Environmental Satellite-R (GOES-R) Concept Design Center Space Segment Team Study, ATR-2002(2331)-1

momentum is also small. In general, slew maneuvers were much more stressing than environmental disturbances, but here they were only a factor of 2 apart. This was because the slew maneuver can be performed slowly, in 30 min, and because the solar pressure torque was high with the one-wing design. The actuator selection, slew needs, and propulsion requirements for momentum dumping are discussed under the attitude control section.

9.2 Design Summary

The pointing knowledge requirement is the same for all configurations because it is derived from the needs of the ABI payload. The latest ACS requirements were taken from T. Kenney, P. Mason and E. Stoneking, GOES-R Study 3, Final Version, October 7, 2002. The document guiding the previous design (A. D. Reth, ACS_req_rev_A'8-9-01.doc, Draft ACS Requirements Document, 8/17/01) listed two sets of requirements, a threshold set and a goal set . The goal then was 1.4 arc-sec, and the threshold was the same 3 arc-sec as now. Table 9.1 summarizes the ADACS design.

The design for Configuration 12, the "A Sat," is considered the nominal case and is described first. The ADACS changes for the other four configurations are listed at the end of this section.

Table 9.1. ADACS Requirements

Design Parameter	Requirement
Stabilization Type	3-Axis Control
Attitude Determination (3σ)	3 arc-sec = 0.0008° = 14 μrad
Attitude Control (3σ)	30 arc-sec = 0.008° = 145 μrad
Slew Requirements	180°/30 min twice per year
Jitter Requirements	Listed as jitter control and knowledge

9.3 Attitude Determination

The backbone attitude sensors are the star trackers. Three arc-sec (3σ) is within the capability of the best star sensors on low Earth satellites that rotate faster (assuming nadir pointing, thus one rotation per orbit) and get more stars through the star tracker field-of-view (FOV). The nadir-pointing geo-stationary GOES Spacecraft stay longer on the same stars, having in effect fewer independent star measurements. They dwell longer on the same pixels of the CCD (charged-coupled device) sensing plane, allowing a slower smoothing of the pixels centroid position. On the positive side, the high geosynchronous altitude reduces the false star hit errors from the South Atlantic anomaly, but this was a smaller effect. The slow rotational rate was a larger effect, but it is still possible to meet the 3 arc-sec spec with highly tested star sensors.

A side analysis during the previous GOES-R CDC study showed that it was even possible to meet the stringent 1.4 arc-sec goal most of the time, and perhaps even all the time. That study was performed to find the attitude determination accuracy with three highly calibrated Ball CT-602 star sensors and a SIRU. The study used a new tool added at that time to the CDC ADACS spreadsheet. This tool (R. M. Dolphus, Simplified Kalman Filters for Control Analysis, Aerospace Technical Memorandum, ATM No. 99(9990)-3, July 29, 1999) calculates the combined effect of star trackers and gyros, and also includes the effect of other position sensors. The tool has subsequently undergone numerous refinements, the main one being the distinction between the star sensor noise and bias error components. The Kalman filter uses the gyro to take out the star sensor noise, but the bias is unobservable to the filter. Using the upgraded tool, it was shown that 3 arc-sec (3σ) is still achievable, including the star sensor bias.

The 2008 technology freeze date permits an even more confident statement than could be made based on the 2004 date used in the first GOES-R CDC study since Ball claims it is testing star sensor prototypes twice as accurate as the currently available models used in the side analysis with the combined sensors tool.

A set of three Ball CT-602 star sensors was selected. They are the best Ball models, and with the usual factory testing and calibration achieve a 3 arc-sec (3σ) accuracy specification. Discussions with Ball indicated they can produce such sensors now, and are working on sub arc-sec prototypes. The CT-602 mass and power were used in the instrument list. The three sensors would be placed on the ABI instrument arranged as the sides of a tetrahedron. They would point South of the orbit plane at an angle larger than 23° to avoid sun impingement. The angle should be no larger than necessary to avoid the sun, to provide the largest possible effective FOV rotation rate and capture more stars. This arrangement provides three-axis accuracy with a small reduction in case of failure of one of the sensors. An optimal geometric arrangement of the star sensor directions provides the highest accuracy in the ABI pitch and roll directions. Taking into account the effective rotation rate and star availability was beyond the scope of the study. Placing the star sensors on the ABI payload minimizes the misalignment effects. A 3 arc-sec budget, with nearly that value already consumed by the sensor errors and Kalman filter performance, allows very little room for misalignments since the two sources are root sum squared (RSS) in the total result. It is difficult or even impossible to keep the on-orbit misalignments (thermal, jitter induced, etc.) between the star trackers and the ABI reference frame small enough (say, less than 1 arc-sec) unless the star trackers are on the ABI, or very near the ABI on a rigid connection.

Other star sensors were mentioned during the first GOES-R CDC study and are worthy of further investigation. The Lockheed AST Mini "1 arc-sec" sensor, basing its high accuracy on more stars in the FOV, seems to have had problems and is not currently supported. Valley Forge Composite Technologies (VFCT) claims it can produce 1 arc-sec star trackers at reasonable prices as star sensors go, i.e., $3M for four units. However, its current star sensors on the International Space Station have lower accuracy, so the high-accuracy models have not been proven yet. The selected star sensors, Ball CT-602, and all the other ADACS instruments are described in Table 9.2.

The gyros selected were Litton Space Inertial Reference Units (SIRU), using four Hemispherical Resonating Gyros (HRG) for each SIRU. A fair amount of space experience has been accumulated with the SIRU in the last seven years, and there have been no reported failures on orbit. This model was selected because of its low noise and drift rate. A single SIRU might have been selected, without a spare, since it is an expensive (approximately $800K) and robust unit. It has four HRGs and in many ways is robust to single-point failures. Any set of three HRGs can measure any of the angular rates in any of three axes. However, to ensure a 10-year mission life and based on discussions about redundancy with the customer, two units were chosen for the design. A single unit selection remains a possible choice. Alternate units that could be considered are the Litton LN-100 IMU based on mechanical rate gyros, or the Litton LN 200 fiber-optic gyros. They have been in use since the early nineties and like the SIRU have low drift and noise. The LN-100 weighs about twice as much as the

Table 9.2. ADACS Equipment List

Instrument	Units	Unit Mass (kg)	Unit Power (W)	NASA TRL	Comments
Fine Sun Sensors	2	0.05	0.1	8	0.017°, 2-axis analog
Coarse Sun Sensors	3	0.08	0.1	8	2°, 2-axis analog, 180° FOV
Star Sensors	3	5.4	10.0	6	3 arc-sec 3σ, Ball CT-602
IMUs	2	4.5	20.0	7	Litton SIRU, 4 HRG low bias
GPS Receivers	2	0.2	4	6	Rockwell NavStrike, light
Reaction Wheels	4	6.5	17	8	Honeywell Constellation HR12, 0.2 N-m, 20 N-m-sec
Thrusters	8	See note	See note	N/A	0.5-m moment arm
Nutation Dampers	3	1	0	9	UCN Aerospace, jitter control
ADACS Computer	0	0	0	N/A	Shared with C&DH
Interface Electronics	N/A	2	10	9	Cables, connectors, boards
Total ADACS	17	57	138	6	Power based on average use

Note: Thruster mass and power consumption are included in the propulsion subsystem.

SIRU, but costs much less (about $100K versus $800K), while the LN-200 has the advantage of being very light.

Sun sensors are used for safe modes and initial acquisition. Two Adcole 12202 fine sun sensors (0.017°) and three Adcole 18394 coarse sun sensors (2°, 180° FOV) were selected. Both models are very light 2-axis analog sensors. The fairly high accuracy 12202 sensors may be useful for star tracker initialization, but the main reason for their selection was the low weight and reputable source. The choice of using only coarse sun sensors for safe modes, since the star trackers can self acquire (though it takes longer), is also viable.

Two GPS receivers were included for orbit determination. This choice was made based on discussions and considering the 2012 launch date. By that date the GPS rf beams will likely be wider, so the number of GPS spacecraft seen from GEO will be higher, permitting GPS-based navigation over larger parts of GEO. Even if GPS is only available part of the time, it permits efficient orbit determination and propagation. At present, few or no GEO satellites use GPS for navigation, partly because the orbital position can be detected accurately enough triangulating with beacons from ground tracking stations, and mainly because of the limited current GPS visibility at GEO. The very light Sandia/Rockwell-Collins NavStrike GPS receiver board was selected (two units). It has been developed for launch and space applications, and Rockwell-Collins is a major producer of military receivers.

9.4 Attitude Control

The standard way to control the satellite attitude to the required accuracy (30 arc-sec = 150 μrad = 0.008°) was with reaction wheels (RWs). The standard set of four RWs arranged in a pyramid configuration was selected. A minimum of three RWs were needed to control momentum in three axes, but four RWs were appropriate for the 10-year mission life, in case one fails. RW failures are rare, but not unheard of. In addition to the electronic components and power supplies, failures can also occur in the mechanical bearings and their lubrication systems.

Unloading the accumulated RW momentum is done with thrusters. An all-thruster system with no RWs would be an undesirable design choice. Even if the thrusters were small enough to be able to control to the required accuracy, the amount of fuel needed for a 10-year life would be prohibitive. The number of firing cycles would also exceed thrusters limits. The RWs control the large daily torque cycle with essentially no fuel penalty. Only the small cumulative component needs to be unloaded with thrusters. Magnetic rods are not a consideration for this type of GEO satellite with a single solar panel and a large daily torque cycle. The rods would weigh more than the fuel needed for momentum unloading during the entire 10-year life.

The RW size was driven by the semi-annual yaw flip. An angular momentum of 14.7 N-m-s was needed to rotate the spacecraft 180° in 30 min. The daily solar pressure accumulation cycle was also substantial, about 7 N-m-s, or nearly half the slew maneuver momentum. The maximum solar pressure torque was approximately 0.00015 N-m. Both the torque and the momentum accumulation were from the solar pressure, with gravity gradient and magnetic torques orders of magnitude below. Had the yaw slew maneuver needs been significantly higher than the daily attitude needs (say, an order of magnitude higher), then one might consider an alternative design with small wheels for environmental disturbances while the slew is carried out with thrusters.

To have an adequate safety margin and to use standard components, wheels larger than the minimum required were used. Honeywell Constellation 6.5-kg wheels, with a 0.2 N-m torque and 20 N-m-s momentum capacity were selected. With this selection, the wheel stays at less than one third of its maximum speed during normal orbit operation since it needs to handle only the 7 N-m-s orbit momentum build-up. There was even more margin since there are four wheels covering the three directions, so on average there are 30 N-m-s of momentum reserve per axis, unless a wheel has failed. Twice a year the wheels incur larger momentum when they perform the yaw flip. With all four RWs, the 14.7 N-m-s slew maneuver uses half the 30 N-m-s per axis reserve. With one failed wheel, there is a 5 N-m-s margin. The wheel specs are listed in Table 9.2.

Thrusters were used for periodic angular momentum dumping. With a 0.5-m moment arm (i.e., distance between the satellite's center of mass and the thruster location), a 1-N thruster force level, and two thrusters per axis with 3-s firings per orbit is required for effective momentum unloading. Again, it was assumed that most of the solar pressure is cyclic, and only 10% of it is cumulative. Otherwise 30-s thruster firings would be needed to unload the momentum. Using a safety factor of 2, a total of 10,000 s of accumulated burn time over the mission life is required. Based on this information, the required propellant mass is calculated by the propulsion subsystem. The approximate fuel amount comes out to 22 kg, based on an I_{SP} of 200 lb/s.

9.5 B Sat Design Overview

The B Sat treated in this configuration does not have the ABI instrument, and the attitude accuracy driver is HES. The pointing requirement is the same, 30 arc-sec, but the knowledge requirement is less stringent, 7 arc-sec instead of 3 arc-sec. The jitter requirements are also less stringent. There was no change in the ADACS sensor's suite from the A Sat to the B Sat because it has the same hardware in weight and power for both cases. The sun sensors and GPS receivers are obviously the same since they don't pertain to high-accuracy attitude determination. The SIRU gyro is the most appropriate unit in both cases. The star sensors, the backbone of the attitude determination system, are the instruments that might be considered for a change to less accurate units. However, this is not the case. The same Ball CT-602s are most appropriate in both cases, 3 arc-sec or 7 arc-sec. In fact, they are the same basic unit for any accuracy below 10 arc-sec. The difference is in the amount of testing these units need to undergo on the factory bench to achieve the given accuracy. There are also significant cost differences. For example, the CT 602 for the A Sat may require 8 weeks of additional testing (over the regular, off-the-shelf unit production), and may cost $2.5M per unit. For the B Sat it may require only 4 weeks of testing, and may cost only $1.5M per unit.

There were no changes in the sensors model numbers in the list of Table 9.2, but there is a change in quality, not captured by the hardware names. This idea was underscored by discussions about the requirements with the customer and the difficulty in choosing between 3 and 7 arc-sec for the attitude determination specifications. Since the sensor selection is not sensitive to that choice, the specification started at 7 arc-sec and then converged to 3 arc-sec, the stated ABI need.

The RW size was increased from 6.5 kg per unit in the A Sat to 8.5 kg per unit in the B Sat to accommodate the larger inertia in the semi-annual yaw flip. This is the only hardware change, and increased the ADACS weight by 8 kg.

Another change pertains to the recommended studies post CDC. The jitter requirements are less severe. It is more likely that the calibrated gyro bias will be small enough to meet the jitter knowledge spec (2 arc-sec over 15 min and 4 arc-sec over 60 min) than in the A Sat case (0.2 arc-sec in 15 min and 0.8 arc-sec in 60 min). However, even in this case, further study is recommended since the non-calibrated gyro bias is not sufficient to meet specifications.

9.6 C Sat Design Overview

The sensors and actuators are the same in the C Sat configuration as in the A Sat configuration shown in Table 9.2, with one minor exception. Due to the smaller inertias, the slew needs can be achieved with slightly smaller reaction wheels, 6 kg per unit instead of 6.5 kg. The total change is a negligible 2 kg savings.

9.7 AB Sat Design Overview

The AB Sat has the same pointing requirements as the nominal A Sat vehicle. Therefore, the sensors are the same. The satellite is much bigger, so the yaw flip requires bigger RWs. That is the only change from the nominal design. The RWs have grown to 14 kg each, from 6.5 kg in the A Sat, for a total weight penalty of 30 kg.

9.8 MEO Sat Design Overview

The MEO Sat pointing requirements were not clearly established, except that they were significantly reduced from the baseline. Per customer input, the 0.03° 3σ accuracy of the best Earth sensor in combination with the SIRU was deemed acceptable for pointing knowledge. The replacement of three star sensors with two Earth sensors has weight, cost, and radiation tolerance benefits. The three Ball CT-602 star trackers weighed 16.2 kg, while two Barnes-Goodrich best Earth sensors, the MMS

13-410, weigh 1.6 kg. The Earth sensor cost is around $300K per unit, versus $1–2M for a highly tested Ball CT-602. For radiation, Earth sensors are much more robust. Star trackers are very sensitive to radiation on the CCD (charged-coupled device) sensing plane, and the optical path to it cannot be shielded.

The RWs were also changed in response to the smaller satellite inertias. Four-kg wheels can handle the yaw flip, so a total of 24.6 kg were saved compared to the nominal design for the A Sat: 10 kg from RWs, and 14.6 from the sensors change.

9.9 Other Design Considerations

Following is a list of assumptions and design issues that are beyond the scope of concept studies.

- The jitter issues are especially important, and further study in that area is recommended in the subsequent design stages to ensure the ABI requirements can be met.

- There is no ADACS-dedicated computer. Mass savings are obtained by sharing the C&DH computer, that is in the RAD 6000 class, or better, and has more than enough throughput for the ADACS software, including star catalog calculations.

- Two kg of cables and interfaces is included for Attitude Control Electronics (ACE).

- Jitter requirements are covered in two ways. One part, called "control" requires 20 arc-sec/s 3σ. The second part, "knowledge," requires 0.2 arc-sec over 15 min and 0.8 arc-sec over 60 min, both values 3σ. The ABI is the driver for these requirements. The jitter control requirements are achievable with the precision rate sensor chosen, but the knowledge part requires further study. The SIRU has a 0.008°/h bias, i.e., 0.008 arc-sec/s. Calibration in the Kalman filter using the star tracker for reference can reduce the bias further. Without those further reductions, the knowledge requirement is not met. 0.008 arc-sec/s implies 7.2 arc-sec/15 min, much more than the 0.2 arc-sec specified. Also, it implies 28 arc-sec/60 min, much more than the allowed 0.8 arc-sec. However, the control part is satisfied. The sensor is within 0.008 arc-sec/s of true rate, and the controller needs to stay within 20 arc-sec/s. There is enough margin for the errors inherent in controller software and actuator nonlinearities. A simulation is recommended, but the prognosis is optimistic to meet controller specifications. For the jitter knowledge, meeting specifications depends on how much of the bias can be reduced by the calibration process. The answer requires further study, and it seems the state of the art is being pushed.

- A study of the jitter sources is also recommended, given the tight jitter knowledge and control specifications. The solar array drive and the RWs are internal jitter sources, and there may be others associated with thruster firings and active cooling of instruments. A jitter study, including a characterization of the jitter sources and of their structural attenuation between the source and the ABI, is needed before deciding with certainty whether active jitter suppression or filtering is necessary. The assumption made here is that there is sufficient damping and frequency separation between the control loop and the jitter sources that the jitter requirements are met passively. Three nutation dampers were included as a placeholder for passive damping.

- Thruster mass and power consumption are accounted for in the propulsion subsystem.

10. Power

Ed Berry

10.1 Overview

The Electrical Power Subsystem (EPS) designs for all GOES Block 5 configurations were based on the following conditions and assumptions:

- Planar, single-wing solar array, one-axis sun tracking, with multi-junction 32% efficient GaAs/Ge solar cells and lightweight Al honeycomb panels
- Li-ion batteries with energy density of 100 W-h/kg, maximum depth of discharge (DOD) of 60% (50% for MEO Sat)
- Spacecraft bus battery-regulated and shunt-limited to 37.5±5 V, same as 2001 GOES-R study
- GEO orbit, 10-year mission, except 19,000-km circular equatorial orbit and 15-year mission for MEO Sat.

10.2 Design Summary

The selection of Li-ion batteries is aggressive because they have little flight heritage as yet although they are under active development and have a fairly large and growing test heritage. Their major attraction is their high energy density, about twice that of NiH_2 batteries. There is now a general perception that Li-ion will ultimately become the standard spacecraft battery type, replacing NiH_2, but the development time necessary to accomplish this is uncertain. It is possible that by 2008 Li-ion technology would not be sufficiently mature to commit to a 10- or 15-year mission. In this case, the alternative technology would be NiH_2, and the battery weights would be about double those calculated in this study.

The 32% solar cell efficiency was based on projections of current technology and estimates of vendors and others engaged in cell research and development. If cell technology in 2008 cannot provide 32% efficiency, then solar array size and mass would be somewhat higher than calculated. Cell efficiencies currently available are ~26–28%.

Trapped radiation dosages were based on the JPL GaAs Solar Cell Radiation Handbook, JPL Publication 96-9, updated to reflect recent radiation test results for multi-junction GaAs/Ge cells. Solar proton dosages were based on the JPL 1991 solar proton event model with an 80% probability of not being exceeded. Solar cell cover-glass thickness of three mils of fused silica was optimal for all configurations (except six mils for MEO Sat), for minimizing solar array mass. Solar cell backside shielding was equivalent to about 18 mils of fused silica.

Major EPS parameters for selected configurations are summarized in Table 10.1. Solar array mass includes allowances for deployment and orientation hardware.

Table 10.1. Power Subsystem Summary

Configuration	**A Sat**	**B Sat**	**C Sat**	**AB Sat**	**MEO Sat**
Solar Arrays					
BOL Power (W)	2817	2703	1320	3985	1526
EOL Power (W)	2354	2258	1103	3330	1024
Solar Array Area (m^2)	8.5	8.1	4.0	12.0	4.6
Solar Array Mass (kg)	46	44	22	65	26
Batteries					
Total Capacity Req'd (A-hr)	99	107	52	144	43
Battery Mass (kg)	38	41	20	55	17
Power Mgt. And Dist. (PMAD)					
Wiring Harness Mass (kg)	60	57	31	80	28
Pwr. Reg. & Cond. Mass (kg)	39	38	20	53	18
Total Power Subsystem Mass (kg)	**183**	**180**	**93**	**254**	**89**

11. Propulsion

Trisha Beutien

11.1 Overview

The propulsion system selected for all configurations was a hydrazine dual-mode system. Hydrazine is used in both the monopropellant attitude control system (ACS) thrusters and in combination with nitrogen tetroxide (NTO) oxidizer, in a Liquid Apogee Engine (LAE) for the orbit transfer maneuvers. The ACS propulsion portion employs a combination of four Aerojet (formerly Primex) monopropellant 5-lb_f thrusters and twelve Aerojet monopropellant 0.2-lb_f thrusters. The LAE is an Atlantic Research 145-lb_f LEROS 1B engine. To achieve a more accurate transfer trajectory, the system is pressure regulated during the transfer and then operates in a blowdown mode for the remainder of the mission.

The four 5-lb_f thrusters, mounted on the same spacecraft surface and oriented in the same direction as the LAE, are used to steer the spacecraft during a LAE burn. It is desirable to also use the 5-lb_f thrusters, with their higher specific impulse (I_{sp}), in the delta-V maneuvers such as station keeping (SK) ones. The requirement that the spacecraft maintains a fixed pointing during most maneuvers, however, only allows the use of the 5-lb_f thrusters when the spacecraft velocity vector coincides with the 5-lb_f thrust vector. Since the spacecraft orientation is unknown, the 0.2-lb_f thrusters are the baselined thrusters for all SK maneuvers. The 5-lb_f thrusters can be used in the supersynchronous orbit disposal maneuver when the payload is assumed inoperative and the pointing requirement (perhaps with the exception of telemetry antennae) is not needed.

An all-bipropellant system (using bipropellant ACS thrusters) would be more efficient in terms of mass but could result in contamination issues with payload optics.* One advantage of the dual-mode system over the bipropellant system is the low thrust level required to support the ACS requirement, achievable with the standard off-the-shelf monopropellant thrusters. The potential exists for mass savings through the use of an electric propulsion (EP) system. The two largest maneuvers in terms of propellant usage (about 90% of total) that can most benefit by the EP are the orbit transfer/insertion and the North South stationkeeping (NSSK).

There are several disadvantages for using the EP in an orbit insertion maneuver. The typical low thrust of the EP requires months instead of days to achieve the GEO from GTO. Besides the associated cost increase and time loss with a longer GTO-to-GEO maneuver, the solar array (SA) will need to be heavier due to a thicker protective coating and larger to account for extra degradation from the prolonged exposure to the Earth's radiation. For most configurations, the large SA is cantilevered to one side of the spacecraft. During an orbit insertion burn, the SA has to continuously rotate to track the sun to meet the high power requirement of the EP. This motion may produce a variable center of gravity and/or variable center of solar pressure, greatly complicating the ACS design and the align-

* Jack, J. R., Spisz, E. W., and Cassidy, J. F., "The Effect of Rocket Plume Contamination on the Optical Properties of Transmitting and Reflecting Materials", AIAA Paper 72-56, San Diego, CA 1972

ment of the thrusters. With the size of the SA being as large as can be practical in some options, the additional increases in size and mass of the SA do not appear feasible.

The EP, if used only in the NSSK, may not need an SA size increase. The operation for most on-orbit maneuvers assumes that most spacecraft payload subsystems can be turned off and the surplus power can be used to fire the thrusters during an EP burn. The requirement by the spacecraft to have all its sensors interrupted as little as possible renders the lengthy NSSK by an EP impractical. The SA size can be increased to enable the NSSK by an EP without disruption to the payload operations. In order to accurately gauge the potential mass savings by using an EP, however, an elaborate scheme of power management may be required for the minimum increase in SA size. Such a power management analysis is out of the scope of a CDC study, and the EP was not explored further.

11.2 Design Summary

The different total vehicle mass for each configuration leads to different required propellant amounts to perform the mission. This, in turn, leads to different propellant and pressurant tank sizes. The number of propellant or pressurant tanks is specified so that the lightest off-the-shelf tanks can be used. The off-the-shelf tanks of a particular size may be lighter than a smaller size, which sometimes results in a dry mass change being inconsistent with the propellant mass change.

Twelve ACS thrusters are the minimum number needed to provide the 3-axis control of a spacecraft. With the relatively large propellant throughput required for the ACS functions, four additional redundant ACS 0.2-lb_f thrusters at selected locations are needed to meet the long-life design. No attempt was made to assess the potential contamination of the optical sensors and SA by the thruster exhaust.

Since all of the configurations are described in the Systems Section, only a summary of the wet and dry mass breakdowns for the 5 main spacecraft configurations (Table 11.1) is included here.

Table 11.1. Propulsion Mass Breakdowns

	Configuration				
	A Sat	**B Sat**	**C Sat**	**AB Sat**	**MEO Sat**
Payload Mass [kg]	519.2	520.8	353.8	698.7	226.7
Bus Mass (w/o propulsion) [kg]	537.3	552.5	364.4	847.0	295.3
Weight Contingency on Payload	25.0%	25.0%	25.0%	25.0%	25.0%
Weight Contingency on Bus	25.0%	25.0%	25.0%	25.0%	25.0%
Thruster Mass Total [kg]	10.1	10.1	10.1	10.1	10.1
Tank Mass Total [kg]	111.6	111.6	72.7	131.5	65.8
Plumbing Mass Total [kg]	14.0	14.0	14.0	14.0	14.0
Propulsion Dry Mass [kg]	135.6	135.6	96.8	155.6	89.8
Operation	Propellant Mass [kg]				
Orbit Insertion	1658.9	1682.4	1133.1	2367.6	857.8
Mission North-South ΔV	376.7	382.1	257.4	537.7	-
Mission East-West ΔV	18.1	18.3	12.3	25.8	5.8
Mission Maneuver ΔV	57.3	58.1	39.1	81.7	12.9
Total accumulated Delta-V time	2.3	2.2	0.7	3.2	3.0

	Configuration				
	A Sat	B Sat	C Sat	AB Sat	MEO Sat
Disposal	7.4	7.5	5.1	10.6	21.0
Residual	42.4	43.0	29.0	60.5	18.0
Pressurant	5.1	5.4	3.5	7.6	2.6
PROP DRY + PROPELLANT	2303.9	2334.7	1576.9	3250.4	1011.0

The only difference between configurations is the amount of propellant they require and the size and number of tanks needed. A summary of the tank information is given in Table 11.2. The larger satellites required a larger amount of propellant, hence larger or more tanks.

Table 11.2. Tank Specifications

	Configuration				
	A Sat	B Sat	C Sat	AB Sat	MEO Sat
Fuel					
Tank Quantity	2	3	2	2	2
Mass [kg]	29	29	17	32	14
Tank Diameter [in]	45	45	39	49	31
Oxidizer					
Tank Quantity	2	2	2	2	2
Mass [kg]	14	14	12	38	12
Tank Diameter [in]	34	34	21 (48 length)	39	32
Pressurant					
Tank Quantity	4	4	2	2	3
Mass [kg]	13	13	8	17	8
Tank Diameter [in]	16.6 (30.6 length)	16.6 (30.6 length)	13.2 (26 length)	17 (35 length)	13.2 (26 length)

11.3 Recommendations/Issues

The NASA Goddard Integrated Mission Design Center (IMDC) propulsion subsystem design differed from the Aerospace CDC design in that it called for a full bi-propulsion system. Thrusters of 22-N were used, which are not in the CDC database. A design was run off-line using Configuration 1 (A Sat-1st Run) with a full bi-propulsion design to allow a comparison with the IMDC design. That is, this design was not iterated with the rest of the CDC, so does not include the effect this would have on other subsystems. The mass breakdown comparison is shown in Table 11.3.

Table 11.3. Mass Breakdown Comparison Between CDC and IMDC Designs

	Bi-prop/Monoprop Combination (318/220 s Isp)	Full Bi-prop (325/250 s Isp)	IMDC Values (325/260 s Isp)
Payload Mass [kg]	398.9	398.9	
Bus Mass (w/o propulsion) [kg]	466.1	466.1	
Weight Contingency on Payload	25.0%	25.0%	
Weight Contingency on Bus	25.0%	25.0%	
Thruster Mass Total [kg]	10.1	28.1	31.7
Tank Mass Total [kg]	87.6	79.0	97.2
Plumbing Mass Total [kg]	14.0	14.0	27.9
Propulsion Dry Mass [kg]	**111.7**	**121.0**	**156.8**
	Propellant Mass [kg]		
Orbit Insertion	1359.2	1250.7	1145.6
Mission North-South ΔV	308.6	195.7	213.8
Mission East-West ΔV	14.8	10.2	5.5
Mission Maneuver DV	46.9	29.7	26.1
Total accumulated Delta-V time	2.1	1.5	29.1*
Disposal	6.1	4.3	5.6
Residual	34.8	29.8	14.3
Pressurant	4.2	3.8	9.5
Propellant + Pressurant [kg]	**1776.6**	**1525.8**	**1492.2**
PROP DRY + PROPELLANT	**1888.3**	**1646.8**	**1649.0**

*IMDC Label is On-orbit ACS - no match in CDC

12. Structure

Kenneth Mercer

12.1 Spacecraft Overview

The structures subsystem was sized based upon an empirical approach found to be reasonable in past studies. Structures mass was derived as a specified fraction of spacecraft dry mass. The appropriate mass fraction was established from a combination of historical data and projected capabilities of technologies. The launch vehicle adapter mass was carried at the systems level as a reduction in the vehicle capability. On dual launch manifests a dual-payload attach fitting (DPAF) adapter mass was estimated based upon current Atlas V PAF designs and included in the structures allocation. Mass contingencies were added at the system level rather than the component level.

Estimates of spacecraft bus dimensions were also developed to enable inertia predictions for ADACS sizing. A maximum dimension is determined from the largest square inscribed within the fairing diameter. In most cases, the spacecraft bus width was dictated by the mounting area required by a given configuration's payload suite. Next, an appropriate height factor was determined to satisfy a specified bus density. A bus density limit of 160 kg/m^3, which does not include payload or solar array mass, was specified. This bound is based upon historical data and is used to address the potential for spacecraft packaging issues. For sizing of the solar arrays, the number of panel segments, each approximately the size of a bus panel, was calculated to meet the total solar array area specified from the power subsystem.

The antenna and magnetometer booms were separately sized to meet stiffness requirements. The design lateral frequencies were greater than or equal to 10 Hz for both booms. Balance mass for center-of-gravity control was selected at 1% of spacecraft dry mass.

12.2 Launch Vehicle Overview

Two launch vehicles were used throughout the study. A Delta IV medium and Atlas V 500.

12.2.1 Delta IV medium

- Fairing height = 622 cm (245 in.)
- Fairing diameter = 457 cm (180 in.)

12.2.2 Atlas V 500

- Fairing height = 488 cm (192 in.)
- Fairing diameter = 457 cm (180 in.)

12.3 Technology Assumptions

The projected structures mass fraction for the current study corresponded to a technology freeze date of 2008. Since use of a commercial bus was preferred, a structures mass fraction of 18% was used.

For this type of large spacecraft, this mass fraction assumes a large amount of composite materials. The structural mass fraction does not include the additional mass from the payload support structure and boom.

Because of the similarity to commercial-type buses with space heritage, the spacecraft structure is a TRL of 7. The Atlas V and Delta IV adapters will be based upon the current adapters; therefore, they are assigned TRLs of 6. A custom adapter is required for all dual launch manifests, resulting in a TRL of 5. A launch isolation system offers a potential mass reduction for the dual vehicle missions, but is not included in the current study. Table 12.1 presents a summary of the technology maturity assessments.

Table 12.1. Technology Assumptions

Spacecraft Structure Mass Fraction	**0.18**
Technology Readiness Levels (NASA TRL)	
Spacecraft Structure	7
Launch Vehicle Adapter	6
Dual Launch Adapter	5

12.4 Design Summary

The A Sat, B Sat, C Sat, and MEO Sat configurations are very similar in their arrangement because they all utilize the top surface of the bus to mount their payloads. The AB Sat contains the full suite of electronics from the A Sat and B Sat; therefore, it requires the addition of a payload module. The payload module is assumed to act as a mini-bus and is sized as a mass fraction of the mounted components. The bus structural mass fraction is reduced to 16% of vehicle dry mass to provide mass credit for the presence of the payload module. Tables 12.2 and 12.3 present the mass summary and structure designs for these configurations.

Table 12.2. Spacecraft Structure Design

	Configurations				
Spacecraft parameters	**"A" Sat**	**"B" Sat**	**"C" Sat**	**MEO Sat**	**"A & B" Sat**
Bus density, kg/m^3	102	93	80	54	92
Effective bus dimension, m	2.5	2.9	2.8	2.5	3
Bus height, m	2.3	1.9	1.5	1.8	2.5
Payload height, m	1.3	1.3	1.3	1.3	4.3
Single bus panel area, m^2	5.6	5.4	4.2	4.5	7.5
No. solar panels per array	2	2	1	1	2

Table 12.3. Structure Mass Results

Mass Breakdown	Configurations				
	"A" Sat	"B" Sat	"C" Sat	MEO Sat	"A & B" Sat
Basic bus structure, kg	215	218	147	110	272
Payload structure, kg	N/A	N/A	N/A	N/A	112
Mechanism, kg	5	10	10	5	10
Balance, kg	12	12	8	6	17
Booms, kg	3	6	N/A	2	7
Total structure, kg	**235**	**246**	**165**	**123**	**418**

For all configurations, it was assumed that on-orbit disturbances are mitigated by ADACS control systems. As a result, no optical bench is utilized to isolate sensitive payload sensors. All configurations exhibit room to fit within the payload fairing. A fitting factor limit of 0.8 was used to leave room for side-mounted components such as the stowed solar arrays. Packaging within the spacecraft is also not a concern since the bus density is well within the allowable limit.

12.5 Recommendations/Issues

Several of the GOES Block 5 sensors require high pointing accuracy. As a result, there is potential for performance degradation due to structural disturbances. It is not possible to reliably predict structural disturbance at the concept design phase; therefore, this analysis should be performed early in the design phases. One potential for combating dynamic disturbance problems is to incorporate an optical bench, but this will incur a mass penalty. Furthermore, it is expected that the inclusion of an optical bench would have a large impact on current configurations because of the already challenging mounting scheme.

On all configurations, the payload suite includes several sensitive optics and sensors. Contamination control of these components should also be addressed early in the program. The key concern is contamination due to off-gassing of composite materials. This is because bus and payload structures are assumed to be mostly composite material.

It is recommended that all dual launch manifests utilize DPAFs. A separate study during the CDC session showed that it was not advantageous to use structure of the inboard vehicle to support an outboard vehicle during launch. In fact, providing a more robust structure on the inboard vehicle revealed a substantial mass penalty.

13. Thermal

Bill Fischer

13.1 Thermal Overview

The thermal subsystem uses relationships between thermal control parameters and the margined orbit average power and the margined spacecraft dry mass to estimate thermal control mass and heater power. This association is based upon data accumulated from many space programs. The thermal designs of the GOES-R spacecraft employ standard commercial satellite thermal control technology. This includes quartz mirrors on radiators with MLI blankets on the remaining external structure. Heat pipes and thermal doublers should be used to spread heat out from concentrated heat sources. Heaters will be required for temperature control at beginning-of-life conditions and during cold periods.

Considerable radiator area will be needed to dissipate the energy generated by the payload. Heat pipes will be needed to move the payload heat to the radiators. The spacecraft bus is a cubic structure with each face of the cube approximately 5 to 6 m^2. The spacecraft orbital orientation has one face of the cube nadir facing. Solar panels extend from adjacent cubic panels to the nadir panel and are oriented toward the sun. The nadir-facing panel is the prime payload sensing equipment location. On each of the payload boxes, radiator area may be located to dissipate a portion of the payload heat. Figure 13.1 shows the amount of area required per watt of power dissipation up to 100 W of dissipation.

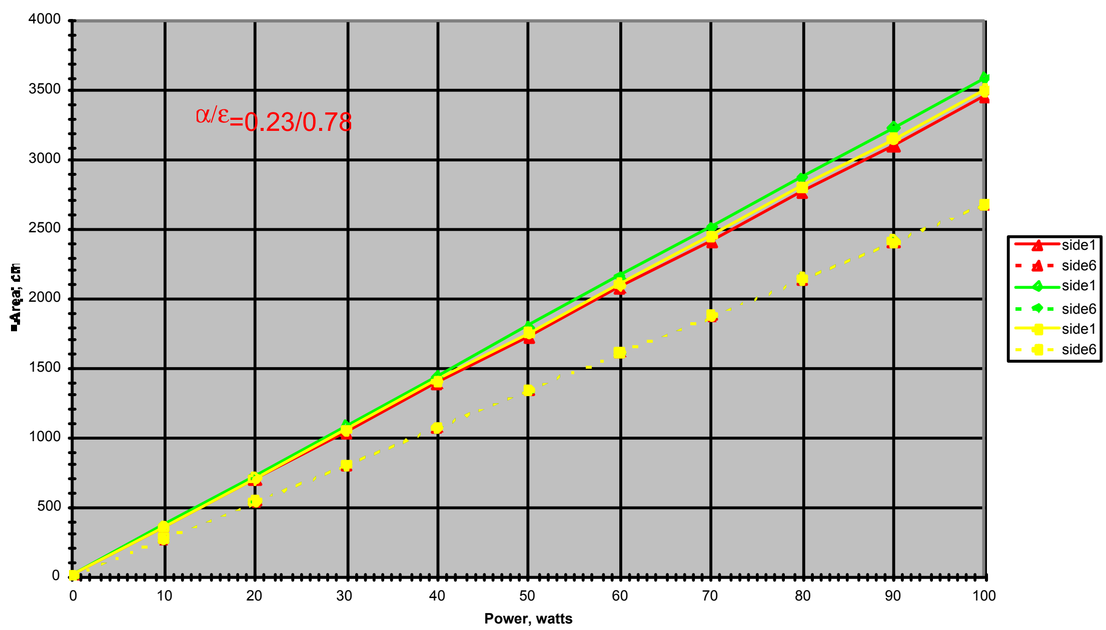

Figure 13.1. Required radiator area for 100 W.

In Figure 13.1, side 1 is the required area for an average face of the vehicle that receives some solar loading during the GEO. Side 6 is the one side of the spacecraft that does not see the sun. For example, this would be the south-facing face of the vehicle when the sun is in the northern hemisphere. Most of the nadir panel is used for equipment location. This leaves approximately five remaining panels to use for radiator locations. Thus, as much as 25 m^2 are available for radiator area. The Advanced Baseline Imager is slated to use a mechanical refrigerator. This device will use a warm-temperature radiator to dissipate the thermal energy generated by the compressor. The original thermal baseline for the Advanced Baseline Imager included a passive cooler or radiator.

A passive cryogenic radiator is a low-temperature, low-power radiator. The radiator area estimated by Donabedian in *Status and Current Technology of Radiant Coolers** is 1.7 m^2 (17,000 cm^2). The passive cooler constrains the vehicle orientation to a 180° flip as the sun passes from the northern to southern hemisphere. This radiator would be located on the side-6 face of the spacecraft. The baseline for the ABI is expected to change to a mechanical refrigerator. A warm-temperature radiator for a mechanical refrigerator radiator has less stringent operational requirements and may not require a seasonal 180° flip. Unfortunately, the refrigerator will require over 100 W of input power to achieve the cooling necessary to cool the ABI.

If the total dissipated power is under 2000 W, as in all the configurations studied, the required radiator area as shown in Figure 13.2 is 5 to 7 m^2. This radiator area is easily distributed within the available area on the spacecraft body.

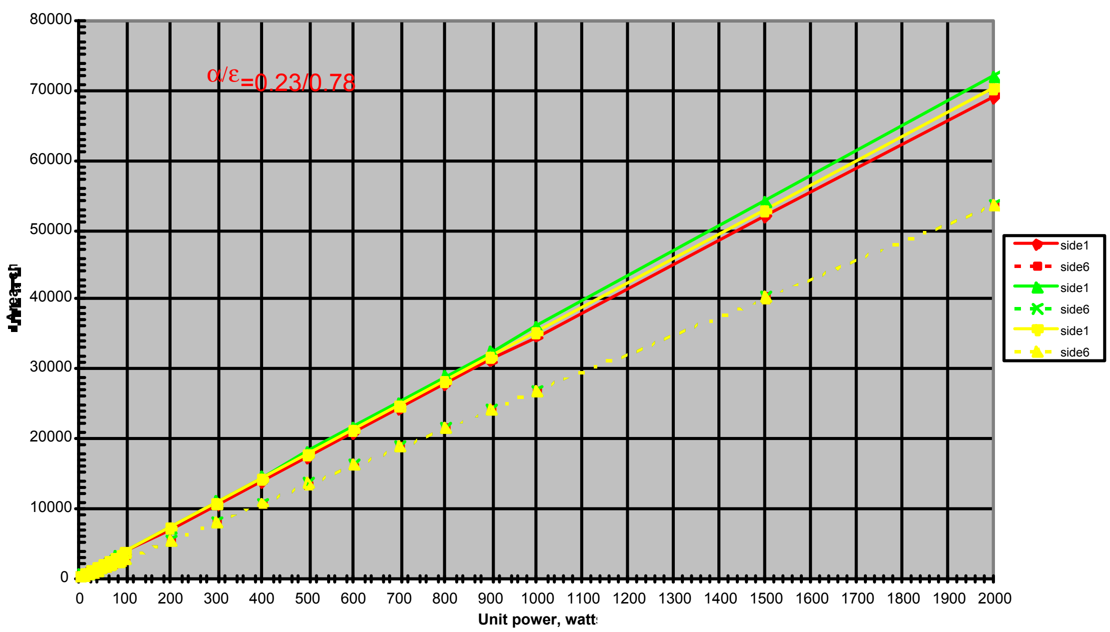

Figure 13.2. Required radiator area for 2000 W.

* Donabedian, Martin, <u>Status and Current Technology of Radiant Coolers</u>, Aerospace Report No. ATR-2001(2331)-2, 24 July 2001.

13.2 Design Summary

The power required for the thermal control system is shown in Table 13.1.

It is immediately obvious that the power dissipated for propulsion; attitude determination and control; command and data handling; and telemetry, tracking, and control are the same for most of the spacecraft designs except the MEO spacecraft. Only the payload, electrical power, and thermal subsystem powers are varying. For all of the bus components, simple passive thermal control techniques such as thermal control coatings and heaters are sufficient. Based on the complexity of payload configuration design, constant conductance and/or variable conductance heat pipes may be necessary to spread the high payload power to body-mounted radiators. The primary thermal concern will be the actual thermal design of each payload element and the location of its radiator.

The design parameters for the thermal mass of the configurations for GOES-R are presented in Table 13.2.

Table 13.1. Thermal Control System Power Requirements

	Power, W							
Configuration	Thermal	Payload	Prop	ADACS	TT&C	CD&H	Power	Total
1A-Sat	191	1119	0.2	138	39	15	20	1522
2B-Sat	195	1118	0.2	138	39	15	24	1529
3B-Sat (option 1)	221	985	0.2	138	39	15	22	1420
4 C-Sat	117	487	0.2	138	39	15	12	808
5 A+B Sat	334	699	0.2	138	39	15	34	1259
6 A-Sat	209	1186	0.2	138	39	15	22	1609
7 A-Sat less Comm	182	866	0.2	138	39	15	16	1257
8 A-Sat with More	234	1286	0.2	138	39	15	24	1736
9 B-Sat	201	1051	0.2	138	39	15	23	1468
10 B-Sat with More	172	732	0.2	138	39	15	17	1114
11 B-Sat less Comm	237	1201	0.2	138	39	15	25	1657
12 MEO Sat	117	440	0.5	114	22	15	19	727

Table 13.2. Thermal Control System Mass

	Mass, kg		
Configuration	Payload Mass	S/C Dry Mass	Thermal Mass
1A-Sat	399	977	28
2B-Sat	388	992	29
3B-Sat (option 1	591	1271	37
4 C-Sat	353	815	24
5 A+B Sat	1003	1701	60
6 A-Sat	458	1065	31
7 A-Sat less Comm	399	925	27
8 A-Sat with More	673	1192	34
9 B-Sat	421	1026	30
10 B-Sat with More	362	877	25
11 B-Sat less Comm	521	1209	35
12 MEO Sat	227	612	18

The primary difference in these configurations is due to different spacecraft mass values associated with the payload configuration. Configuration 5 represents the maximum payload mass with the combined A and B satellite payload configuration. The thermal requirements are significantly lower for the remaining payload configurations.

13.3 Recommendations/Issues

The vehicle's orbit average power provides the best indication of the thermal control heater power requirements and thermal control total weight. At low spacecraft power dissipations, thermal control subsystems are extremely simple, relying on existing structure for radiator area and using bulk spacecraft temperatures to keep equipment within a nominal temperature range. As the spacecraft power increases, dissipated power densities will increase, leading to added complexity in the thermal control subsystem. Thermal doublers and heat pipes may be required to spread localized power dissipation, dedicated radiators may be needed to reject high heat loads to space, and additional heater power may be necessary to keep equipment within allowable temperature. Considerable mass may be added to the spacecraft design. The mass of these systems are roughly 8 to 9 % of the spacecraft dry mass as compared to the standard thermal control system of 3 to 5 % of the spacecraft dry mass. The thermal subsystem can be refined once a design layout is selected and iterated between payload power, design integration, structures, and thermal.